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NASA Contract NAS 5-302

PROJECT APOLLO

*A Feasibility Study of an Advanced
Manned Spacecraft and System*

FINAL REPORT

VOLUME II. SYSTEM CONSIDERATIONS

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Prepared for:

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May 15, 1961

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APPENDIX

SC-A — Notes on Staging

Staging for the family of operational missions will include release of the primary booster stages as a general requirement. In addition, for a modular spacecraft the mission module and on-board propulsion will be released prior to re-entry. Finally, in the likely event that specialized abort rockets are used, it is a logical requirement that they be released after their capability is no longer required. Staging requirements and possibilities for further investigation are discussed below.

SPECIALIZED ABORT ROCKETS

A number of possibilities prevail for satisfying the propulsion requirements for abort (including retro-capability for re-entry). These requirements could be met partially by the on-board propulsion available for the nominal mission, but will require some additional specialized propulsion system. Since the requirement for immediate escape is eliminated after release from the third stage, it is logical to release the final stages of abort rocketry with the third stage. Trade-off of abort rocket requirement and weight results in the apportionment of the total on-the-pad abort capability among a series of abort rockets. These rockets are released in sequence during the primary boost phase. Logically, these releases will occur with first-, second-, and third- stage primary booster releases. Specific details on this are given in Chapter III of this volume.

STAGING TO RE-ENTRY

Prior to re-entry, the re-entry vehicle configuration must be established. Generally, this will involve the release of mission module and on-board propulsion, including all supporting structure, tankage and fairing. The solar power collector, the extended antenna, and the thermal radiator would be released with the structure and fairing.

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SC-B — Definition of Vehicle Subsystems, Crew, and Field-Support Functions

The definitions given here provide the meaning applied to the categories listed in the Functional Status and Operation Chart. The functions listed are summary and general by intent, and in some instances, represent several of the APOLLO task groupings combined into one category. The blocks in the Functional Status and Operation Chart list only the unique or more significant functions as they occur, and do not include the general or ordinary (e.g. intra-crew communications, computation, and maintenance).

INSTRUMENTATION AND COMMUNICATIONS

1. Accumulation, processing, and transmission of vehicle, crew subsystem, and scientific instrumentation data, including status and response. Voice and telemetry.
2. Receipt, processing, distribution, and display of voice and command data from the ground, such as navigational parameters, beacon signals, guidance or control commands, and ground-computed diagnoses of vehicle equipment malfunctions.
3. Intra-crew voice communications. Mission (scientific) and vehicle (subsystem performance) instrumentation.

NAVIGATION AND GUIDANCE

1. Sensing of navigational data and computation to determine vehicle position, velocity, and flight path.
2. Determination of vehicle attitude and motion.
3. Compute, direct, and regulate the action of the flight control functions.
4. Appropriate displays and controls for crew participation.

FLIGHT CONTROL

1. On-board propulsion for basic maneuvers and course corrections in space (including attitude hold); includes fuel storage, distribution, and regulation.

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2. Subsystem for vernier reaction control of attitude.
3. Control surface devices for aerodynamic lift and drag.

ELECTRICAL POWER

1. MAIN -- Long-term functioning in space, including charging of the auxiliary system. Probably solar powered, but could include nuclear supplies.
2. AUXILIARY -- Short-term use during launch, landing, and recovery, emergency, main supply failure, routine maintenance. Includes batteries and fuel cells.

CREW SERVICES

Includes all environmental control and life support functions, such as:

1. Eating, rest, and waste disposal facilities.
2. Air-conditioning, acceleration couches, spacesuits, and radiation protection.
3. Tools and materials for on-board maintenance and repair.
4. Display of all required data.

LANDING AND RECOVERY

Means with which to achieve retardation and terminal maneuver following re-entry into the earth's atmosphere. Impact attenuation, landing gear, brakes and stabilizers, high-seas landing and floatation devices and their deployment. Equipment to support survival, location, and retrieval of crew, payload, and vehicle.

CREW

Three men responsible for directive, monitoring, and response activity during the entire mission, including override control capability of most normal operations. Primary control responsibility for certain operations, including scientific data gathering, communications, and on-board maintenance and repairs. Integrated, where possible, into subsystem functions such as navigation, computation, and guidance.

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LAUNCH AND LANDING SERVICE

Final assembly, fueling, and launch facility operations, including the necessary logistic, power and control functions. Principal and auxiliary (backup) landing-site facilities to accommodate all normal landings and post-touchdown service, including possible field deceleration aids.

COMMUNICATIONS AND TELEMETRY

Central transceiver of network and all ground stations. Master and remote ground control panel and display (medical, subsystems, trajectory). Telemetry reception and reduction. Computation (including tracking data) and command.

TRACKING, SEARCH, AND RETRIEVAL

Central and all ground radar tracking stations. Trajectory monitoring and supplementary guidance (instrumentation and beacon service). Normal landing approach control. Locations of emergency landing sites and recovery operations.

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SC-C — Modal Analysis

Background

One of the most important problems in the design of manned spacecraft is that of achieving a satisfactorily high level of operational availability⁽¹⁾ of all subsystems within size, weight, and other mission limitations.

The General Electric Company Technical Military Planning Operation (TEMPO) has evolved a technique called modal analysis which was successful in:

- (1). Predicting the operational availability of the Polaris fire control subsystem.
- (2). Establishing a measure of maintainability so that the maintainability of the fire control system can be specified, predicted and quantitatively measured.

In accomplishing the foregoing, there was evolved a quantitative measure of subsystem performance considering all modes of operation and the effectiveness of each mode. This measure was called subsystem value.

A further look into this type of analytical approach for determining the operational availability of complex electronic equipment in general⁽²⁾ has lead to the conclusion that "the technique of modal analysis is a tool applicable to evaluation of those modern complex systems which are capable of operating in various states or modes with different degrees of task capability for each mode. Thus, in case of a failure, the entire task capability of the system may not disappear, but the system may keep some measure of its task capability either through judicious redundant design, sympathetic design, or by operating in a lower

- (1) Operational Availability is defined as good time, or "up" time, during the mission divided by total mission time. Bovaird, R., and Zagor, H., "A Systems Approach to Predicting and Measuring Polaris Fire Control System Operational Availability," GE Report RM 59TMP-57, 7 December 1959.
- (2) Bovaird, R., "An Analytical Technique for Determining the Operational Availability of Complex Electronic Equipment," GE Report RM 59TMP-58, 11 December 1959.

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(or alternate) mode until maintenance can be affected to restore the equipment to its prime mode. Such multi-moded systems are becoming rather common in our present complex technology in contrast to the more simple earlier equipments which generally possessed one operating mode."

While not minimizing the results achieved on the Polaris application, it is recognized (1) that the application was purposely limited to one subsystem, (2) the mathematical treatment was based on a subsystem essentially already designed, and (3) the small number of design changes "recommended" by the mathematical treatment were primarily of the pure redundancy type.

Spacecraft Application

Modal Analysis — with its subsystem value measure — as applied to manned space vehicles must be carried much further. First of all the complexity is much greater. Many subsystems and the myriad of interactions among them must be considered.

Secondly, to design for the maintenance management concept in order to maximize the probability of man's survival and mission success, a new measure of evaluation must be developed. This new measure is called SYSTEM WORTH, and it will enable us to determine the relative worth of each and every subsystem to the accomplishment of the mission and manage the maintenance action accordingly.

The expansion of the subsystem value measurement technique, coupled with the development of the system worth evaluation tool, will permit the systems designer to effectively control the extent of sympathetic design necessary to ensure the best design integration compromise.

We shall develop this new mathematical reliability tool and show its relation to the sympathetic design process by presenting the following steps:

- (1). Power supply example - to illustrate:
 - (a) The advantage of knowing - and using this knowledge - when a subsystem(s) is required to be used in the mission profile.
 - (b) The effects of one phase of sympathetic design; in this particular instance, an extreme case of redundant design.
- (2). A discussion of some quantitative factors entering into system design criteria.
- (3). Lunar reconnaissance example - to illustrate:
 - (a) Again, the advantage of knowing when a subsystem must be in use during a mission.

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- (b) The concept of modal operation.
- (c) Probability of occurrence of each modal operation.
- (d) Effectiveness of each mode in accomplishing the specific task of the subsystem.
- (e) Modal value of each possible mode of operation and the concept of subsystem value.
- (f) Essentiality of each subsystem in contributing to the success of the particular mission phase underway.
- (g) System worth measurement for maintenance management.

Development of System Worth Measurement - Power Supply Example

Let us assume we have the same subsystems as before; namely,

- (1). Life-support subsystem
 - (a) Electronic and electrical elements
 - (b) Mechanical, hydraulic, chemical elements
- (2). Communications subsystem
- (3). Radar subsystem
- (4). Guidance and navigation subsystem
- (5). TV subsystem
- (6). Infrared subsystem
- (7). Maintenance control (monitoring and display) subsystem

Let us further select the power supply in each subsystem.

In the case of the life support system, a power supply failure lasting more than a few seconds might be intolerable, a situation which would probably necessitate at least one redundant power supply in the equipment. However, in the case of the radar guidance and communication systems, a much longer downtime might be permissible if it did not come at a critical point during the cruise; and the TV system might be dispensed with altogether without completely aborting the mission.

Assume that the space mission is to last for seven days and the re-entry phase of the mission requires six of the seven subsystems to be operational.

Our example considers three possible design approaches and the reliability figures for each approach are tabulated below:

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(Failure rate per power supply is assumed to be 0.00004 per hour; therefore, the probability that any one power supply will survive the seven day cruise, assuming an exponential failure distribution, is 0.9933.)

| <u>Case</u> | | <u>Probability of required 6 power supplies being available for re-entry</u> |
|-------------|--|--|
| I. | Conventional type design - each power supply designed individually - no redundancy $(0.99332)^6$ | 0.9606 |
| II. | Sympathetic design - integrated block approach; i.e., each power supply completely interchangeable one with another | 0.9991 |
| III. | Conventional type design - each power supply designed individually <u>and a</u> <u>redundant hot spare</u> provided for each of required 6 subsystems. | 0.99973 |

As a practical matter, of course, it would be unfeasible to fully comply with either Case II or Case III. However, by judicious use of the sympathetic design approach, it is feasible that many subassemblies and components can be designed with switching as required so as to obtain a degree of cross utilization among the various subsystem power supplies with this cross utilization resulting in a probability figure somewhere between 0.9606 and 0.9991.

System Design Criteria - Quantitative Factors

If any system configuration is to be optimized in the strict sense of the word, a meaningful quantitative design criterion must be established whereby alternative system configurations can be adequately evaluated. This type of measure must include the following four factors:

- (1). The inherent reliability of each replaceable module, subassembly, assembly and the subsystem in the system. This is the mean time to failure of the subunit.
- (2). The predictability of failures (via marginal checking or statistical techniques based on wearout characteristics). To the extent that failures can be anticipated, their effect can be largely counteracted.

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- (3). The criticality of each sub-unit in terms of its effect on subsystem performance effectiveness.
- (4). The worth of each subsystem to the total system at each point in time during the cruise. This value will be dictated by the mission profile.

Development of System Worth Measurements - Lunar Reconnaissance Example

INITIAL ASSUMPTIONS

Following is a simplified example of the application of the modal analysis technique to a hypothetical space vehicle.

Let us assume that the mission is to reconnoiter the Moon and map its surface on the far side. While the vehicle is performing the actual mapping operation on the moon's far side, its earth communication subsystem is idle.

Let us further assume that a wide bandpass amplifier in the earth communication system has been deliberately over designed so that it can function to a degree as a preamplifier for the radar transmitter in the reconnaissance (mapping) subsystem. Note that during the period of communication subsystem idleness, this particular wide bandpass amplifier is available as a replacement (or redundant) part in lieu of the radar transmitter preamplifier.

Furthermore, the overdesigned amplifier can be switched to the reconnaissance subsystem to parallel radar transmitter preamplifier operation for the duration of the mapping operation. At the termination of the mapping phase, the overdesigned amplifier is returned to the communications subsystem for use on the return trip to earth.

DEVELOPMENT OF MODES OF OPERATION

Let: A represent the radar transmitter pre-amp;
B represent the radar transmitter amplifier;
C represent the communication broad bandpass amplifier.

A + B now represents a portion of the surveillance subsystem suitable for analysis.

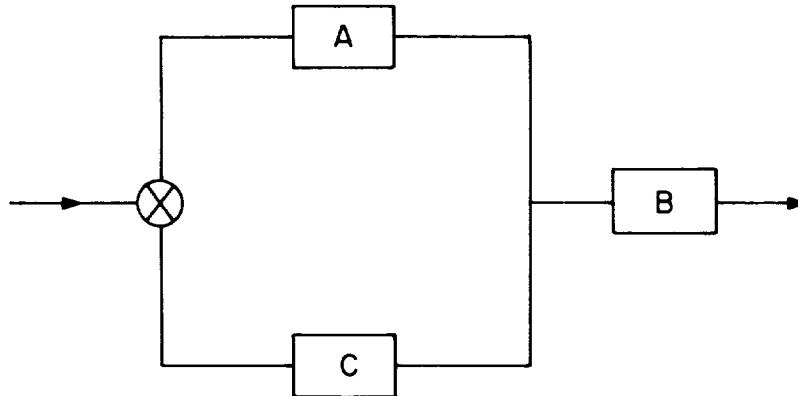
Assume:

- (1) If A works, and B works the system is fully effective.
- (2) If A fails, A + B is worthless.
- (3) If A fails, C can be switched in.

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- (4) If C works and is switched in, there is a degradation in the quality of the map obtained.
- (5) If A and C both fail, the system is worthless
- (6) If B fails, it will fail in such a fashion as to yield a map which still has some value.

The following schematic illustrates the situation.



We can identify five modes of operation for the radar assembly, each of which has a finite probability of existing at any randomly selected point during the mission. In the symbology at hand:

| | | |
|----------------|---|---|
| M_1 (Mode 1) | = | A + B working |
| M_2 (Mode 2) | = | A only working (B failed) |
| M_3 (Mode 3) | = | C + B working (A failed) |
| M_4 (Mode 4) | = | C only working (A failed and B failed) |
| M_5 (Mode 5) | = | All other combinations (subsystem failed) |

PROBABILITY OF OCCURRENCE OF EACH MODE OF OPERATION

The probability assigned to each mode is dependent on the operational availability of the individual subassemblies involved. In turn the operational availabilities are dependent on

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the reliability (mean time to failure) and maintainability (mean time to return to service) of A, B, and C. ⁽¹⁾

Assigning the following operational availability numbers:

- a = 0.9, the operational availability of A
- b = 0.8, the operational availability of B
- c = 0.85, the operational availability of C
- d = 0.99, the operational availability of the switch, X.

(Note that 1-a, 1-b, etc. pertain to the operational unavailability of a, b, etc. respectively.)

The modal probabilities are then:

| | | | | |
|----------|---|------------------------------|---|-------------|
| $P(M_1)$ | = | $a \cdot b$ | = | 0.72 |
| $P(M_2)$ | = | $a(1-b)$ | = | 0.18 |
| $P(M_3)$ | = | $(1-a) \times c \cdot b$ | = | 0.067 |
| $P(M_4)$ | = | $(1-a) \times c \cdot (1-b)$ | = | 0.017 |
| $P(M_5)$ | = | $1.0 - P(M_1)$ | = | 0.016 |
| | | | | <hr/> 1.000 |

MODAL EFFECTIVENESS

Having determined the probability of occurrence for each mode, the next step is to assign an effectiveness number to each mode, this number being based on how well each mode performs the assigned task.

- E_1 = 1.0 (the system is operating in its primary, or most effective mode; A & B working)
- E_2 = 0.5 (the radar transmitter, B, has failed so as to give a map of only 50% normal quality; A only working)
- E_3 = 0.9 (a degradation in map quality of 10% results from using the communications amplifier as a radar transmitter pre-amplifier; C & B working, A failed)

(1) See, Bovaird, R., "An Analytical Technique for Determining the Operational Availability of Complex Electronic Equipment," GE Co. TEMPO Report RM 59TMP-58, 11 December 1959.

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| | | |
|-------|---|---|
| E_4 | = | 0.45 (the communications pre-amp is being used with a malfunctioning transmitter; C only working; A & B failed) |
| E_5 | = | 0.00 (no useful subsystem function) |

We are now in a position to compute the "value" of each mode.

MODAL VALUE

If we define modal value, V , as equal to modal effectiveness, E , times the probability that the mode will occur, $P(M)$, then $V(M) = P(M) \cdot E$, and for each mode of operation we have the following modal values:

| | | | | | | |
|----------|---|--------------|---|----------------|---|--------------|
| $V(M_1)$ | = | $P(M_1)E_1$ | = | (0.72) (1.0) | = | 0.72 |
| $V(M_2)$ | = | $P(M_2) E_2$ | = | (0.18) (0.5) | = | 0.09 |
| $V(M_3)$ | = | $P(M_3) E$ | = | (0.067) (0.9) | = | 0.06 |
| $V(M_4)$ | = | $P(M_4) E$ | = | (0.017) (0.45) | = | 0.008 |
| $V(M_5)$ | = | $P(M_5) E$ | = | (0.016) (0.0) | = | 0.0 |
| V | | | = | | = | <u>0.878</u> |

The total value (V) of the radar subsystem is thus seen to be 0.878 during the lunar mapping phase of the mission.

The 0.878 figure statistically is a dimensionless figure and standing alone means very little. However, when several different figures representing several different subsystem design configurations are developed, the system designer now has a method of directing and evaluating integrated subsystem sympathetic design efforts.

The obvious first thought might be to maximize the occurrence of those modes of operation with the high effectiveness values. However, in many instances, practical implementation will cause the systems designer to concentrate on increasing the modal probabilities of the relatively low effectivity modes.

ESSENTIALITY NUMBERS

If each subsystem retained the same relative priority of operation throughout the mission profile, we could assign these subsystems permanent priority ratings and use the modal value measurement to optimize our vehicular system design. Furthermore, these same priority ratings could be used in determining sequence of maintenance management action.

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However, the relative importance of each subsystem to man's survival and the successful accomplishment of the mission changes at various points along the mission profile. For instance, referring to our earlier example of a lunar reconnaissance mission, we have hypothesized a mission profile by subsystem for this vehicle as illustrated in Figure C-1.

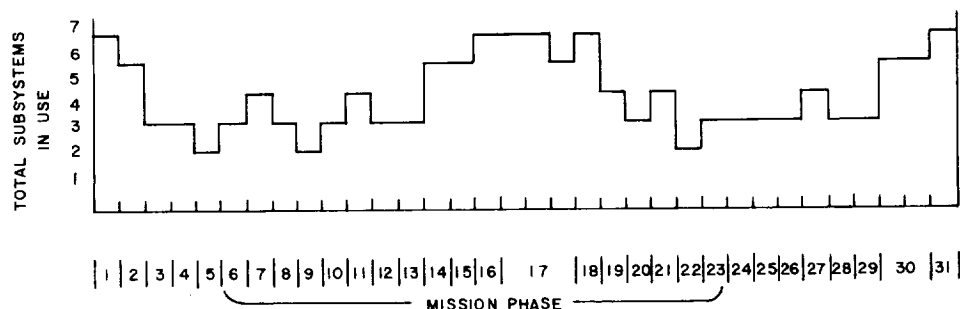
Note that only two of the subsystems, life support and maintenance management, are required continuously throughout the journey. On the other hand, as mentioned previously, the communication subsystem is not needed during the actual lunar mapping operation on the Moon's far side. Furthermore, it is not absolutely vital during many other portions of the flight. Likewise, the TV subsystem is vital only during certain limited time intervals of this particular mission and is not required during a greater portion of the flight.

Therefore, we now have to examine each phase of the mission and for each phase assign essentiality numbers (Y) to each individual subsystem. This essentiality number is an arbitrary number which denotes the relative importance each subsystem plays in accomplishing a particular phase of the mission.

Adapting a zero-to-one scale for essentiality numbers, we can say that if a subsystem's failure to function during a particular phase of the mission is likely to cause a catastrophic mission failure (as in the case of the life support system at any time), the essentiality of that subsystem during that time interval is 1.0. If, on the other hand, a subsystem is not required during the phase under analysis, its essentiality is 0.0. Subsystems whose failure to function degrades mission performance but is not catastrophic to mission success will have essentiality numbers between 0.0 and 1.0.

As an example of essentiality number application, let us assign numbers to the seven subsystems previously hypothesized for our lunar recon vehicle. However, before doing so let us make the following definition:

A mission phase is defined as that interval of time during the mission profile in which there is no change in any essentiality number assigned to a subsystem.



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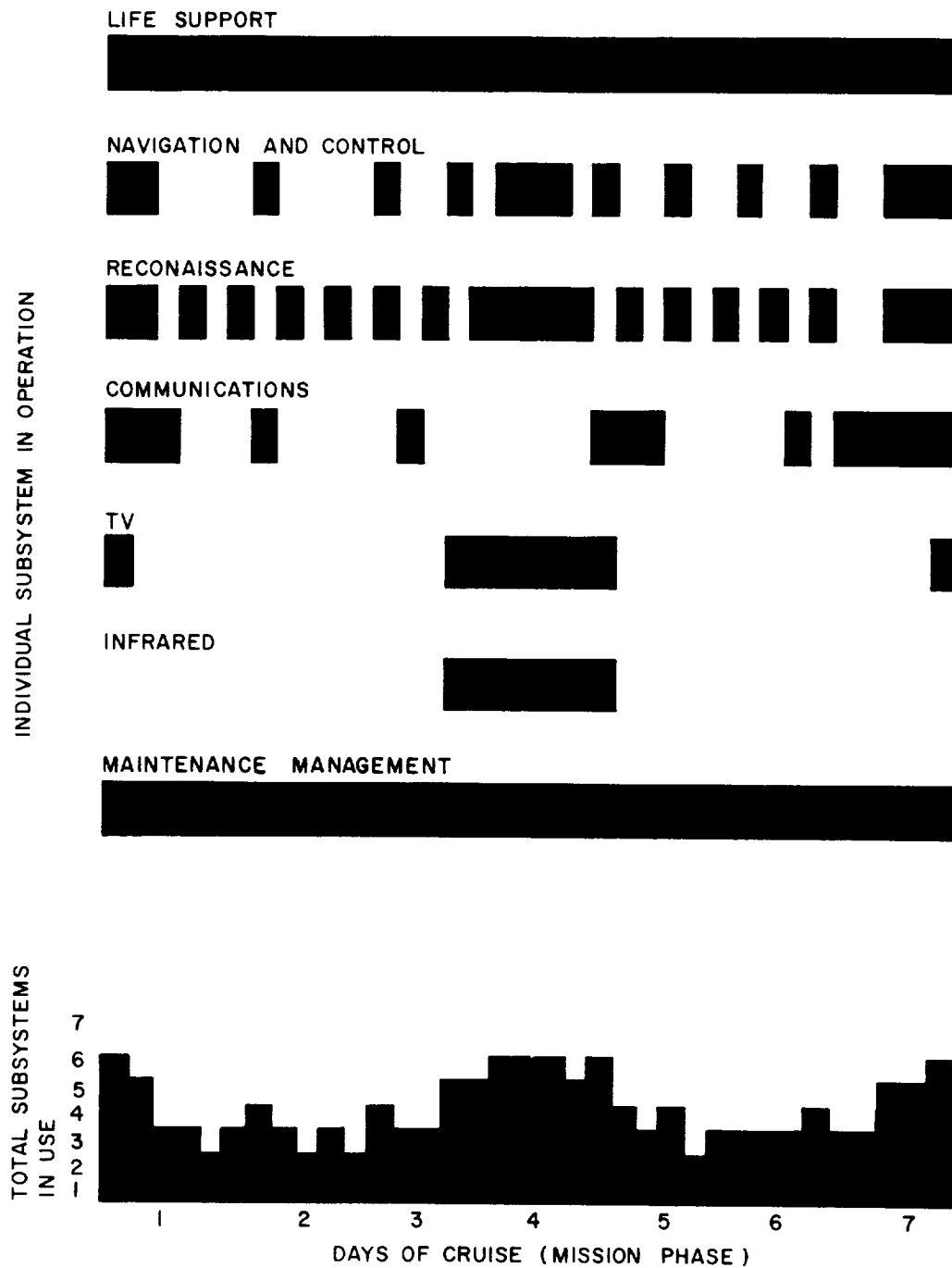


Figure C-1. Hypothetical mission profiles for seven subsystems
one-week lunar reconnaissance

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The foregoing chart shows the mission profile broken down by mission phase. If we now refer to the following table we find that we have arbitrarily assigned hypothetical essentiality numbers to a majority of mission phases.

With the exception of the life support subsystem, the essentiality numbers assigned to each subsystem vary considerably throughout the mission profile. The systems designer who has the task of integrating sympathetic design techniques among the various subsystems must now have an additional mathematical evaluation tool beyond the subsystem value (V) previously developed.

| SUBSYSTEM | MISSION PHASE | | | | | | | | | | | | | | | | | | |
|-------------------------------------|---------------|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|-----|
| | 1 | 2 | 3 | 4 | 5 | 6 | 7 | 8 | 9 | 10 | 11 | 12 | 13 | 14 | 15 | 16 | 17 | 18 | 19 |
| Life Support | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 | 1.0 |
| Nav and Control | .9 | .9 | 0.0 | 0.0 | 0.0 | 0.0 | .9 | 0.0 | 0.0 | 0.0 | 0.0 | .9 | 0.0 | 0.0 | .8 | 0.0 | .8 | 0.0 | .8 |
| Reconnaissance | .4 | .4 | 0.0 | .9 | 0.0 | .9 | 0.0 | .9 | 0.0 | .9 | 0.0 | .7 | 0.0 | .9 | 0.0 | .9 | .9 | .9 | 0.0 |
| Communications | .7 | .7 | .8 | 0.0 | 0.0 | 0.0 | .6 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | .9 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | .9 |
| TV | .2 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | .6 | .8 | .5 | .8 | .3 |
| Infra-red | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | 0.0 | .9 | .6 | .7 | .6 | .5 |
| SMAC (Maintenance Management) | .8 | .8 | .9 | .8 | .9 | .8 | .8 | .8 | .9 | .8 | .9 | .8 | .8 | .8 | .7 | .7 | .6 | .7 | .7 |

Three cases can be used to illustrate the need for such a mathematical tool:

- (1) In the initial design stages, the systems designer has to integrate the amount of sympathetic design among amplifier functional families found in the communications, navigation and control, TV and reconnaissance subsystems. If he had only one mission phase to consider, he could assign his essentiality numbers - or rank subsystems in order of importance - and optimize the subsystem value of the most important ones.

However, in this particular lunar recon example, the system designer has 31 different mission phases to consider, and he needs some means to satisfactorily inter-relate the various subsystem values with the degree of essentiality for each subsystem throughout the duration of the mission.

- (2) At the last moment, a decision has been made that because of certain factors a reduction in payload must be accomplished, and the systems designer has been given the task of reducing component circuitry some ten pounds. The systems designer needs some method of evaluating the best way to reduce weight resulting in the least amount of degradation of mission survival and/or success.

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- (3) At the last moment the systems designer finds that he can add a couple of hot spares to his payload circuitry. Again, he needs some positive evaluation technique to help him select the best compromise.

SYSTEM WORTH

Considering a single mission phase, if we take the essentiality number (Y) of a given subsystem and multiply it by the subsystem value (V) previously determined, we now have a quantitative measure which we call subsystem worth (W). This is a statistical method of measuring a subsystem's contribution to the success of that particular mission phase.

After proceeding further to calculate the worth of each subsystem in every phase of the mission, we can, by summing up the individual subsystem worths per phase, arrive at an over-all vehicular system worth for each phase. We have now established a basis for evaluating quantitatively alternative subsystem design configurations and arriving at the best compromise.

OPTIMIZING SYSTEM WORTH

However, in order to permit the systems designer to arrive at the best integration decision, there must be some means of optimizing this systems worth measure so that the chain effect of sympathetic design changes can be quickly evaluated. Certain analytical techniques from the fields of production economics, statistics, and operations research can be employed to facilitate the search for an "optimal" equipment configuration.

Should the relationships involved prove linear, a simple linear programming model might solve the problem of what combination of sympathetic design effort and cross utilization of functional subunits will maximize system worth within the weight and volume constraints.

However, since linear relationships are unlikely, another possible approach is to construct a non-linear response surface from which can be derived the marginal rates of substitution of the various factors (spares versus redundant circuits, for example). An optimal allocation of resources can then be determined by analysis of these rates or the method of Wilde or Box-Wilson (method of steepest ascent) can be used to search for a maximum system worth on the response surface.

EFFECT OF WEIGHT AND VOLUME CONSTRAINTS

Rather rigid constraints on the weight and volume of equipment carried on the spacecraft may prohibit the achievement of any pre-assigned value of "system worth" over all missions.

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Instead, the goal will be to maximize total system worth (over the entire cruise) within the weight and/or volume limits.

The decision as to whether to add a redundant circuit in one subsystem or another will then depend on the contribution to system worth per pound (or per cu/ft) contributed by that circuit in the different configurations. The same sort of analysis may be applied to spare modules; and the method here might be to keep adding sequentially the redundant circuit, test circuit or spare module which will contribute the most to system worth until the weight limit is reached.

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SC-D — Mathematical Basis and Methodology for Preliminary Reliability Estimates

The notation $R(t)$ is used as the reliability of a piece of equipment (component, system, etc) of time (t) ; i. e. the probability of no failure occurring in the period from the beginning of the operation up to the time t is $R(t)$. Since probability is a statistical concept, the probability calculus is used in the calculation of reliability, and statistical techniques are employed in the measurement of reliability.

A major premise made in the mathematics is that complex systems, which have been debugged and which consist of many components operating under flight environmental conditions, are assumed to have, for all practical purposes, a constant failure rate. Expressed mathematically, this means that the probability of no systems failure occurring within a given time interval (t) is an exponential function of that time interval.

This exponential function can be mathematically described by means of a single parameter "m" called the mean time between failures (MTBF) of the system:

$$R(t) = e^{-\frac{t}{m}} \quad (1)$$

By plotting this curve $R(t)$ against the time to failure (t) , one obtains the survival characteristic of the system, see Figure D-1. The values of $R(t)$ range from one for $t=0$ to zero for $t=\infty$ and the area under this curve equals:

$$A = \int_0^{\infty} e^{-\frac{t}{m}} dt = -m \left[e^{-\frac{t}{m}} \right]_0^{\infty} = m$$

or, in general, the mean time between failures is equivalent to the area under the reliability curve:

$$m = \int_0^{\infty} R(t) dt \quad (2)$$

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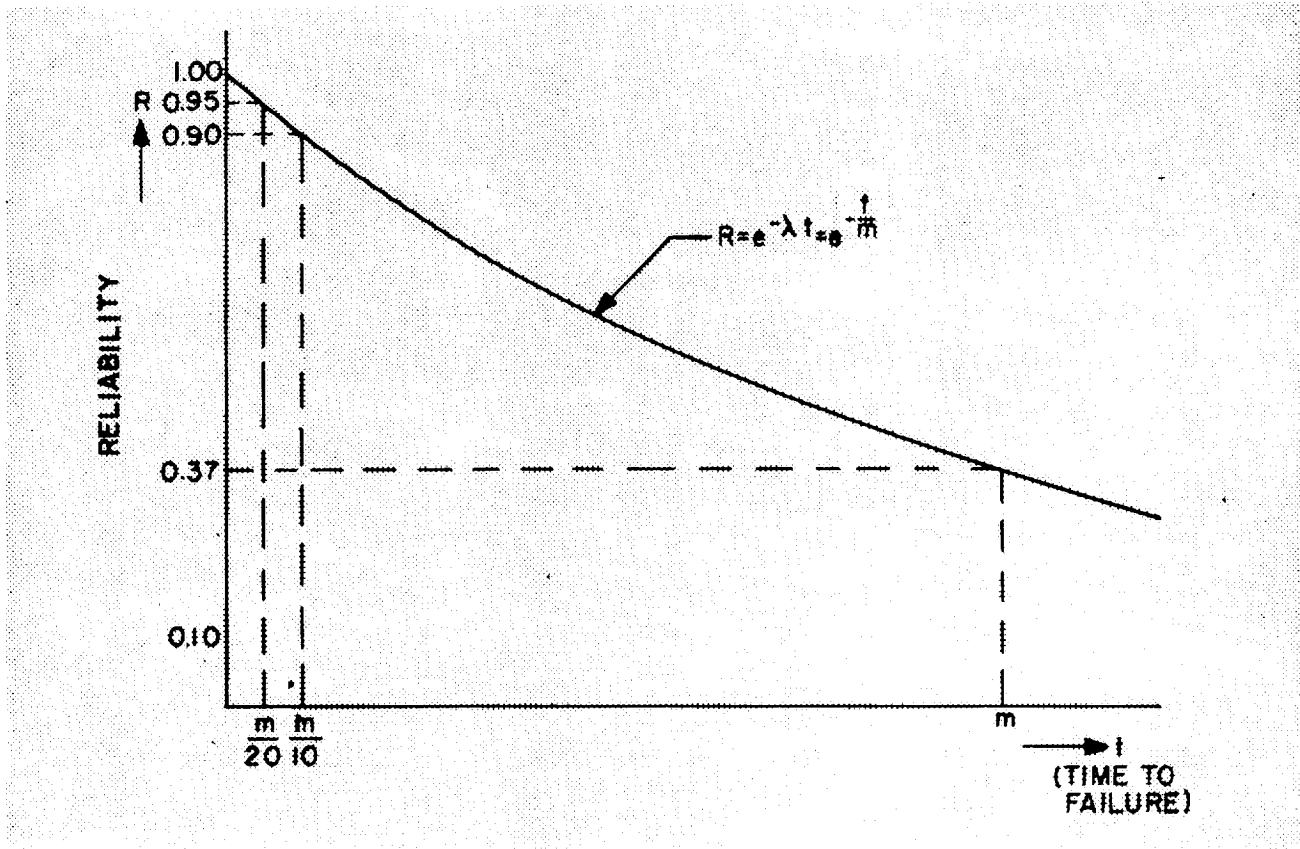


Figure D-1. Reliability Curve for $\lambda = \text{Constant}$

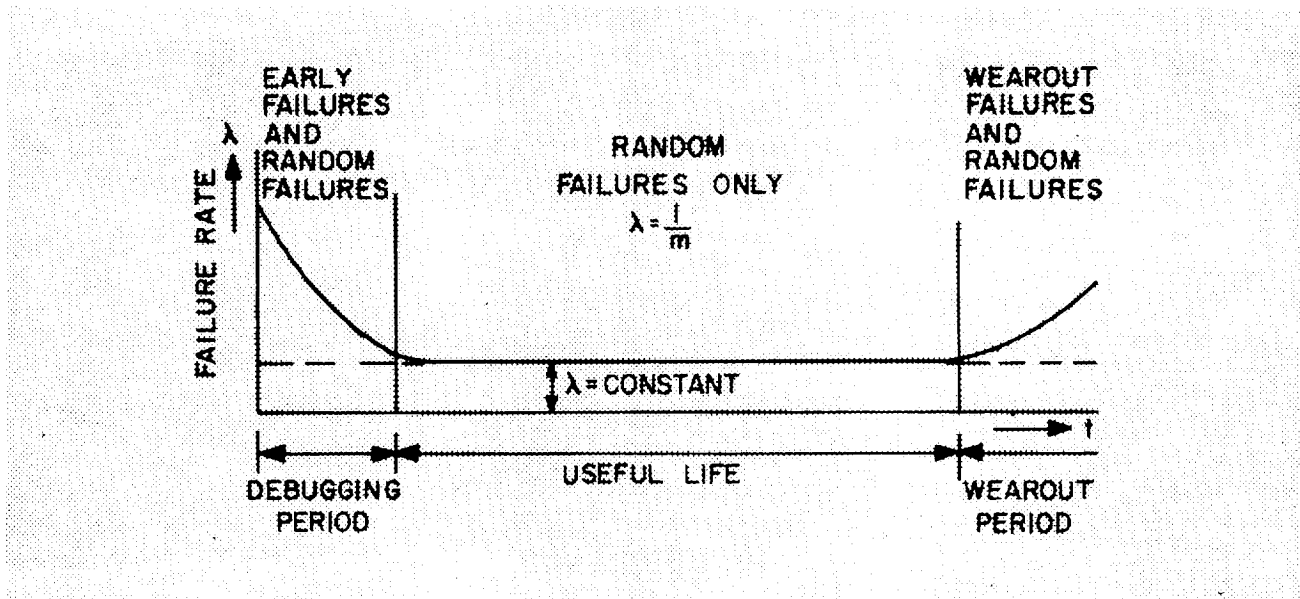


Figure D-2. Failure Rate Curve

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Seen physically, $R(t) = e^{-\frac{t}{m}} = e^{-1} = \frac{1}{e} = 0.37$, which means that for an operating time equal to the mean time between failures ($t=m$) there is a 37% probability of failure-free operation of the system.

Failure Rate

The failure rate of a system is defined as the number of failures per unit time, such as per one operating hour (but often appearing as percent per 1000 operating hours). The failure rate is not constant for the entire life of an equipment.

When making a graph of the failure rate for the whole life, the failure rate is usually high at the beginning and will usually show a decreasing tendency. It would usually become constant in a comparatively short time, i.e. when the system reaches a state of essentially constant failure rate. (See Figure D-2.) The period of decreasing failure rate is called the debugging or "burn-in" period. Infantile (early) failures, which are inherent in the new product and are due to manufacturing or wiring errors and material weaknesses, usually show up rather soon when the product is put into operation. When the curve straightens and runs essentially parallel to the abscissa, the failure rate is approximately constant. The product has reached its period of useful life where only chance failures occur at random. Later the failure assumes an increasing tendency, degradation failures beginning to appear as a consequence of age (wear-out) when the equipment is reaching its "rated life".

The constant failure rate λ in the useful life period is the reciprocal of the mean time between failures (chance failures):

$$\lambda = \frac{1}{m} \quad (3)$$

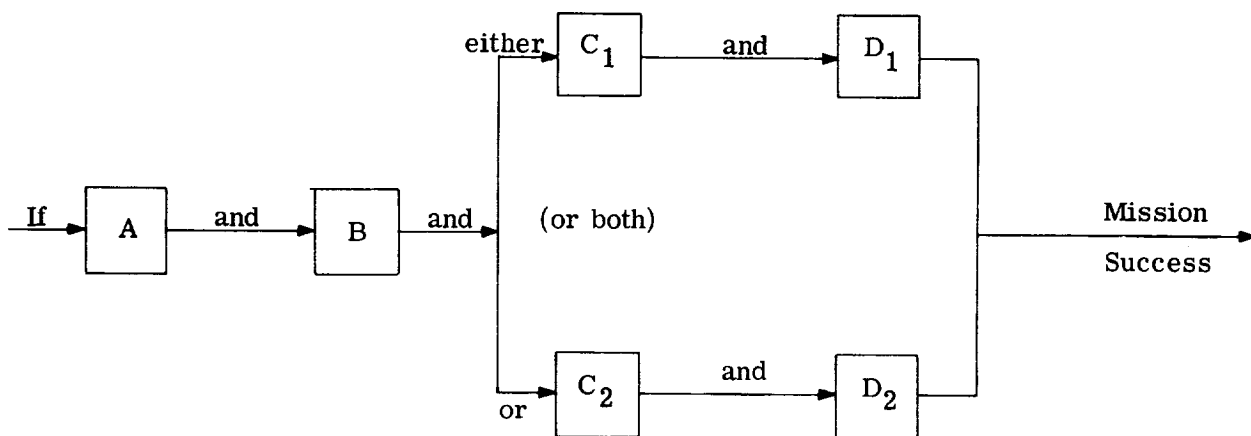
The Mathematical Model

The basic approach to the analysis is made from the failure rate point of view. The procedure presented herein is limited to components which are essentially assemblies capable of subdivision into parts for analytical purposes. An example of a component which can be subdivided into parts for such analysis is an amplifier circuit. On the other hand, a battery, although generally treated as a component cannot realistically be subdivided into its parts for this purpose. The steps listed below provide an orderly procedure for the performance of an analysis for a component, and provide the basis for subsystem and system estimates.

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The reliability block diagram is a representation of the functional relationship, from a reliability point of view, of the part to the overall component performance. Within each block, representing a part or group of parts is indicated what must not occur to the part, in terms of mode of failure, in order that the component perform its intended function. The reliability block diagram plays an important role in the reliability design analysis by pictorially describing the mission of the component. An example is given below of a reliability block diagram of a fictitious component where the blocks represent functional parts or circuits and are lettered for purposes of simplicity.



The mathematical model relates the probability of success of the parts or circuits which comprise the component to the probability of success of the component. In general, the probability that the component will perform as required is equal to the product of the probabilities of success of its parts. When certain parts are redundant within the component, then the probability of success will depart from the product rule. The model for the reliability or probability of success of the fictitious component given above is developed by the following steps:

1. In verbal terms: The component will succeed if; sub-units A and B operate and either C_1 and D_1 or C_2 and D_2 operate or C_1 and D_1 and C_2 and D_2 all operate.
2. In Boolean terminology the model would be given as:

$$R_{\text{Component}} = P \left(A \cap B \cap \left[(C_1 \cap D_1) \cup (C_2 \cap D_2) \right] \right) \quad (4)$$

Where: R = Reliability

P = Probability

\cup = Symbol for "or"

\cap = Symbol for "and"

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3. The above equation reduced to the mathematical model:

$$R_{\text{Component}} = R_A \cdot R_B \cdot \left[1 - (1 - R_{C1} \cdot R_{D1}) (1 - R_{C2} \cdot R_{D2}) \right] = R_A \cdot R_B \cdot 1 - (1 - R_C \cdot R_D)^2 \quad (5)$$

where $R_{C1} = R_{C2}$ and $R_{D1} = R_{D2}$

Of importance, here, is that the reliability of the component is equal to the product of the reliabilities of its parts; redundant parts being reducible to equivalent parts for purposes of the analysis.

Computational Methods

The exponentially distributed reliabilities can be combined, for the series or non-redundant situation, in the following manner:

$$R_s = e^{-\lambda_1 t_1} \cdot e^{-\lambda_2 t_2} \cdot \dots \cdot e^{-\lambda_n t_n} = e^{-\sum_{i=1}^n \lambda_i t_i} \quad (6)$$

Where:

R_s = Total or combined reliability of the non-redundant parts

λ_i = Failure rate of the i^{th} part.

t_i = Operating time of the i^{th} part

\sum = Symbol for summation

When all parts are time-sensitive or cyclically-operated, and the mission duration is the same for all parts, then the above equation becomes:

$$R_s = e^{-t \sum_{i=1}^n \lambda_i} \quad (7)$$

where:

t = total time

For those parts which are redundant, an equivalent failure rate may be derived and subsequently factored into the series expression above. The reliability of a part in redundancy is the probability that at least one element operate. This can be expressed alternatively as:

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$$R_{wr} = (1 - \text{all elements fail.})$$

$$R_{wr} = 1 - (1-p_1) (1-p_2) \text{-----}(1-p_n) \quad (8a)$$

$$R_{wr} = 1 - (1-e^{-\lambda_1 t}) (1-e^{-\lambda_2 t}) \text{-----} (1-e^{-\lambda_n t})$$

where $p_1, p_2, \text{-----}$ etc. = probability of success of individual elements, and $\lambda_1, \lambda_2 \text{-----}$ = their respective failure rates.

With elements of equal failure rates: $\lambda_1 = \lambda_2 = \lambda_3 \text{-----} \lambda_n = \lambda$,

$$R_{wr} = 1 - (1-e^{-\lambda t})^n = 1 - (1 - R_{\text{single element}})^n \quad (8b)$$

While equations 8 gives the reliability for a working redundancy, the instance of a sequential redundancy is accommodated by:

$$R_{SR} = R_1 + (1-R_1) R_2 + (1-R_1) (1-R_2) R_3 \text{-----}$$
$$R_{SR} = \sum_{i=1} R_i \cdot \prod_{j=0}^{j=i-1} (1-R_j); R_0 = 0 \quad (9)$$

where:

R_i = Reliability of the i th substitute unit

R_j = Reliability of the j th failed unit

\prod = symbol for product

In the event that one or more "one-shot" devices are employed, these reliabilities can be estimated from past experience, vendor data, analogous part or equipment types or intuitive engineering expectation for the design. These reliabilities can be combined product-wise with the exponentially distributed reliabilities determined by equation (6.)

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SC-E — Some Preliminary APOLLO Subsystem Reliability Estimates

Some of the reliability estimates generated during the course of the APOLLO study are presented here. These estimates have only slight quantitative basis in fact, although they do serve as preliminary qualitative indications of the level of achievement which the preliminary designs present. The estimates are based on pessimistic failure rate data; as such, they are generally conservative and should not be construed as indicative of the reliabilities expected in design. The estimates are for the 14 day lunar orbit mission.

Communications

The reliability estimate of the vehicle-borne communications system is based on the NASA guideline requirement for near-continuous communication with the vehicle. The final system design provides both voice and telemetry capability. Near-continuous coverage is construed to mean voice and telemetry communications provision on an uninterrupted basis if possible.

The system has 3 possible operating modes:

- a) 2 kmc
- b) 250 mc
- c) 400 mc emergency

The 250 mc system operates to an altitude of 8000 nmi. At this point in the outbound trajectory, the system is switched to the 2 kmc deep space mode and remains in that mode until the vehicle reaches an altitude of 8000 nmi on the return trajectory. At this point the system is switched to the 250 mc mode and remains in that state until touch-down. The 400 mc emergency system can be used at any point in the mission as conditions require. The system is described in detail in Volume VII, Chapter III of this report.

The failure rates for the major subsystems are as follows:

| | | |
|-------------------------------|---|--------------------------------------|
| λ 250 mc | = | $.0407 \times 10^{-3}$ failures/hr |
| λ 2 kmc | = | $.0407 \times 10^{-3}$ failures/hr |
| λ encoders | = | $.013744 \times 10^{-5}$ failures/hr |
| λ ant. servo | = | $.025 \times 10^{-5}$ failures/hr |
| λ multiplex, timer | = | $.0124 \times 10^{-3}$ failures/hr |
| λ 400 mc | = | $.0407 \times 10^{-3}$ failures/hr |

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The mathematical model set up to define the reliability of the system is based on the following statement:

The probability of successful communications is:

The probability that the 250 mc system does not fail and the 2kmc system does not fail and the digital equipment does not fail and the 2kmc antenna servo does not fail or the emergency system does not fail.

The Boolean expression for the above statement becomes:

$$R_{\text{Total}} = R_{250\text{mc}} \cdot R_{2\text{kmc}} \cdot R_{\text{digital}} \cdot R_{\text{servo}} \cup R_{\text{emergency}}$$

The 250 mc system operates for a total period of 2 hours.

The 2 kmc system operates for a total period of 240 hours. This figure was determined as follows:

Total mission time = 14 days = 336 hrs.

Lunar blackout (7days in orbit) = -94 hrs.

250 mc System operates: -2 hrs.

Total 2 kmc ON-Time 240 hrs. (also servo "on-time")

The digital equipment operates for the total mission (336 hrs.)

It is assumed that the 400 mc system may have to operate for the total mission (336 hrs.).

The reliability of the subsystems then follows from the combination of the failure rates and the operating time.

$$\begin{aligned} R_{\text{servo}} &= e^{-0.025 \times 10^{-5} (240)} = .9999 \\ R_{250\text{ mc}} &= e^{-0.041 \times 10^{-3} (2)} = .9999 \\ R_{2\text{ kmc}} &= e^{-0.041 \times 10^{-3} (240)} = .9902 \\ R_{\text{digital}} &= e^{-1.374 \times 10^{-5} (336)} = .9953 \\ R_{\text{emergency}} &= e^{-0.041 \times 10^{-3} (336)} = .9863 \end{aligned}$$

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$$\therefore R_{\text{Total}} = R_{250\text{mc}} \cdot R_{2\text{kmc}} \cdot R_{\text{digital}} \cdot R_{\text{servo}} \cup R_{\text{emergency}}$$

$$R_{\text{Total}} = 1 - \left[1 - R_{250\text{mc}} \cdot R_{2\text{kmc}} \cdot R_{\text{digital}} \cdot R_{\text{servo}} \right] \left[1 - R_{\text{emergency}} \right]$$

$$R_{\text{Total}} = 1 - \left[1 - .9854 \right] \left[1 - .9863 \right]$$

$$R_{\text{Total}} = 0.9998$$

This estimate is conservative in that it was assumed that both voice and telemetry contact are required on a continuous basis.

Navigation, Guidance, and Control

The navigation, guidance, and control subsystem reliability hinges upon the successful performance of the gyro stabilized platform, the attitude control system, the celestial sextant or manual navigation aids, and the re-entry flight control system. Attitude control propulsion and flight control propulsion, as well as radio backup, are considered elsewhere.

It is considered that single redundancy of complete inner gimbals of the stable platform will provide better than double redundancy due to the duplication of types of equipment in the gimbal itself. Further reliability is assured by the relatively low (approximately 50 hr) duty cycle of the platform equipment. The platform is shut down when its function may be carried by other more reliable navigation and control equipment. The overall mission reliability of the platform system has been estimated as 0.9994 by using those techniques.

The overall reliability of the attitude control system is dependent upon the reliability of its individual attitude sensors (e.g. Stable Platform, Celestial Sextant, sun sensors), the guidance computer, and the rate gyros. Due to the multitude of attitude references available, this reliability approaches 1.0. The computer reliability is 0.965. By carrying complete redundancy in rate gyros their reliability for the mission can be shown to be 0.998.

The reliability of the non-redundant Celestial Sextant for the mission is 0.9967. By carrying complete redundancy for the guidance computer key modules, its mission reliability can be shown to be 0.965.

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The entire navigation, guidance and control system reliability is approximately 0.96. This figure relies upon the ability of the crew to replace or repair inoperable components.

Landing System

The landing system is taken here to include retardation (parachute) and shock attenuation. The detailed description of this system is given in Volume VII. The entire system is essentially a "one-shot" arrangement. Various reliability values assigned to operational components have been established from past recovery system testing on the RVX-2A program. Where reliability or failure rate values are not available, conservative estimates have been made and are so noted in the table below:

| <u>COMPONENT</u> | <u>RELIABILITY</u> |
|--|--------------------|
| 1. Control Unit (with mechanical timer and pyrotechnic switching elements) | 0.95 |
| 2. "g" Switch | 0.93 |
| 3. Power Supply | 0.92 |
| 4. Parachute | 0.98* |
| 5. Ejection Charges and Squibs | 0.95* |
| 6. Impact Bag (mechanical failure) | 0.99* |
| 7. Pressure vessel and Plumbing | 0.92 |
| 8. Explosive valve | 0.82 |
| 9. Altitude switch | 0.95* |
| 10. Parachute reefing cutters | 0.88* |

* Estimated

A reliability block diagram is shown in Figure E-1; it should not be interpreted as a physical representation of electrical or mechanical connections. To successfully complete the landing without exceeding the crew's limitations the following functions must occur in sequence.

1. One of two arming switches in redundancy must arm the control unit and power supply.
2. At least one of two altitude switches must close to activate the control unit.
3. At least one of two control unit/power supply combinations must operate to provide proper sequencing to the system.

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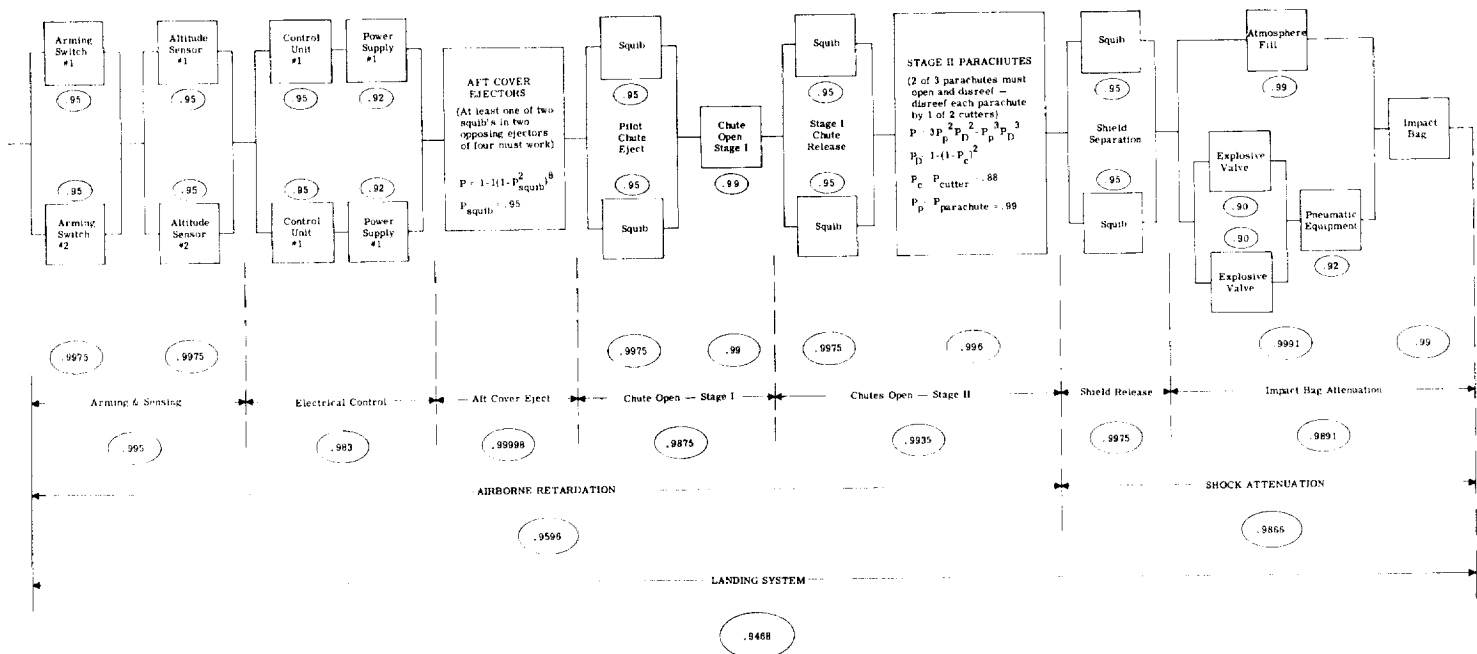


Figure E-1. APOLLO landing system - reliability estimate

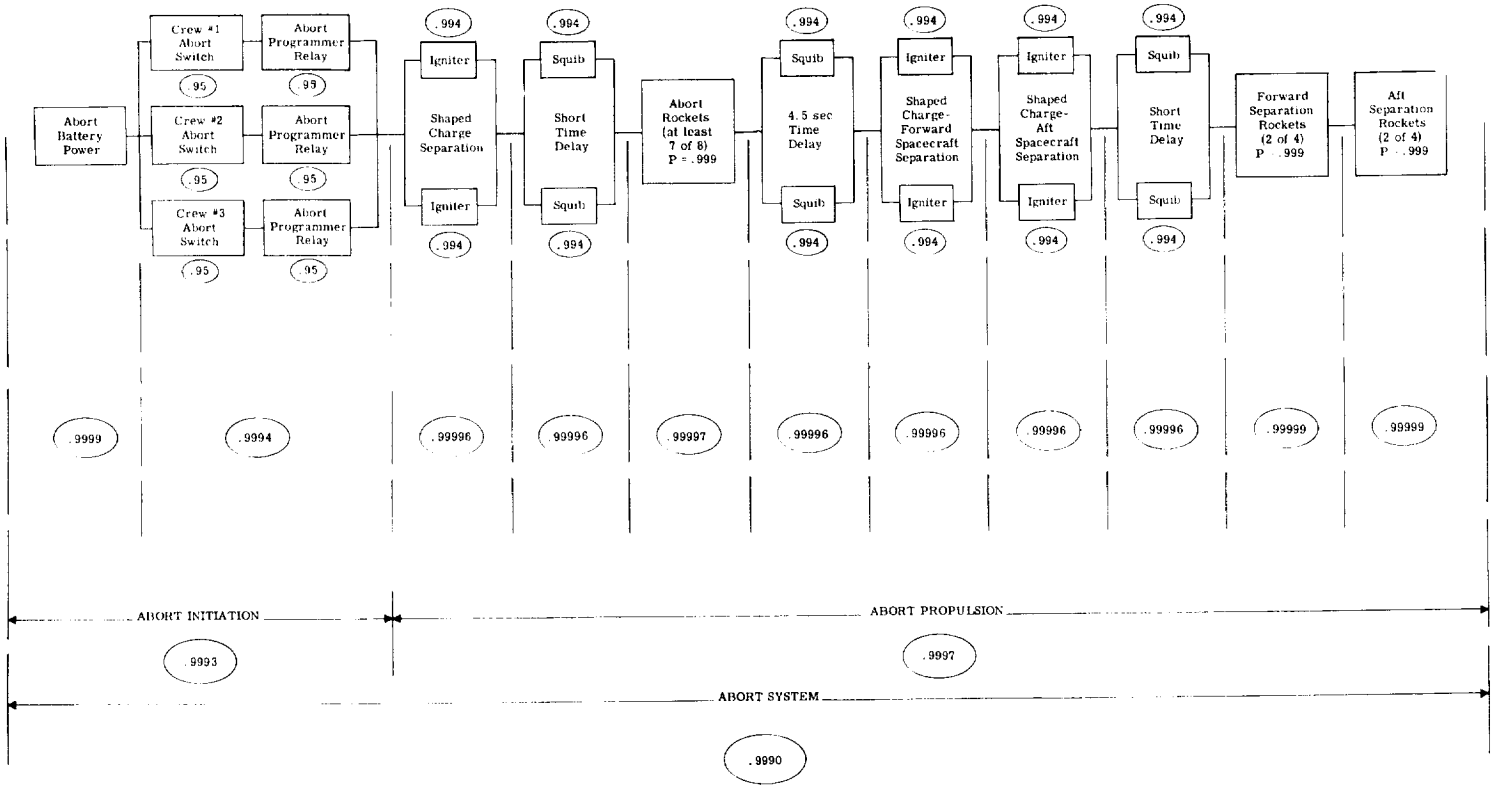


Figure E-2. On-pad and high q abort system reliability estimate

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4. The aft cover must be ejected by firing at least one of two squibs in two opposing ejectors of four.
5. Pilot chute ejection must occur through firing of at least one of two squibs.
6. The stage I chute must deploy, and:
7. Release by means of at least one of two squibs.
8. At least 2 of 3 parachutes must open and disreef. Disreef each parachute by at least one of two squibs.
9. Separate the shield by firing at least one of two squibs.
10. The impact bag must be automatically filled by the atmosphere or by at least one of two explosive valves triggering the pneumatic equipment to fill the bag.
11. The impact bag must have structural integrity.

The several-fold combination of estimates in Figure E-1 develop an overall reliability estimate of 0.9468. Again, it must be stated that this is at least a crude prediction. It should be noted here that no dependency on the crew has been factored into this sequence. Where it becomes possible in the design, an improvement in the predicted reliability should become apparent.

Abort System

The reliability estimate for the abort system under the most severe conditions (on-pad and high q) is given in Figure E-2. The details of the system are discussed in Chapter III of this volume.

On - Board Propulsion

The table below gives the reliability estimate in terms of mission accomplishment from the propulsion standpoint only. This data is extracted from Appendix A, Volume IV.

| <u>Phase Description</u> | <u>Estimated Reliability</u> |
|-------------------------------|------------------------------|
| First Midcourse Connection | .99385 |
| Lunar Insertion | .98215 |
| Lunar Exit | .99390 |
| Second Midcourse Connection | .98442 |
| Re-entry Vehicle Separation | .999975 |
| Re-entry Vehicle Pull Control | .999410 |

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CHAPTER I

MISSION ANALYSIS

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CHAPTER I

MISSION ANALYSIS

Analysis of the APOLLO mission has been based on the NASA Project APOLLO guidelines (RFP 302), Project Mercury considerations, and Saturn booster information. The principal results of this analysis are embodied in the mission profile. Another aspect of this analysis is the selection of landing sites. The results obtained to date in these two areas are presented in this section.

1.0 Mission Profile

The complete mission profile will time-define (a) the mission phases, (b) the respective operations of the space craft subsystems, crew and supporting ground complex, and (c) the environmental profile. In final form, mission profiles will cover not only the operational missions, but also the abnormal modes and developmental flights. The intent of the early mission profile is to provide preliminary design guidelines in terms of environmental constraints, system composition, and subsystem functions and operations. During later phases of the program, including operational, the updated mission profile forms the basis for equipment test and checkout, and may provide an outline for the countdown procedure.

The present report emphasizes the operational missions. Most of the elements of developmental flights can be derived from the operational mission elements; however, these flights are not specifically covered in this report. The general definition of the abnormal, or emergency, flights is given through a keyed relationship with the operational missions in this section; they are given detailed and separate treatment in Chapter III of this Volume.

The mission is established through definition of its component phases. The basic missions described here are the earth orbit, lunar orbit, and circumlunar. The lunar landing is also included in general terms, but not detailed at this time. The mission phases and

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associated timing given here constitute a realistic model, adequate for preliminary design, but, by no means do they reflect detailed accuracy. The latter will be possible only after a succession of iterations, into which are factored the refined trajectory, configuration, and functional element data. The phases have been selected on the basis of some effect or requirement uniquely associated with each. For this reason, we see timing intervals of days intermixed with those of seconds and overlapping among phases.

Subsystem functional requirements are defined, and their operations uniquely tied to the individual mission phases. An environmental profile for the basic missions is also included.

1.1 PHASE DEFINITION

The family of operational APOLLO missions is shown in summary form in Figure I-1-1. Major mission regimes are indicated with reference time marks. These times are defined from liftoff at t_0 . Lunar reconnaissance mission alternatives are the circumlunar pass and the lunar orbit. Although not a present requirement, the option of a lunar landing, exploration, and launch is indicated. Another possible alternative is a direct boost into a lunar trajectory or by way of an earth-parking orbit. A final alternative, by implication, is the early operational earth orbit which will have requirements well within missions 2, 4, or 6 with the exception of third-stage booster separation after injection into orbit and possibly added requirements for ground tracking of extended orbit time. These variations are summarized in Figure I-1-2 with the more important phase time intervals noted. These phases and time intervals are defined in Figures I-1-3 through I-1-9, inclusive. The basic criterion for selection of these particular phases for presentation here has been the unique requirements of each, whether functional or environmental. For this reason, certain time intervals are measured in seconds, others in days.

During assembly and pre-launch (Figure I-1-3), four phases are of particular significance. Factory sub-assembly and transport will extend over a period to within about two months of the launch date. This would appear to be a minimum time for field activities prior to launch; however, it is suggested here as an objective for operational APOLLO missions which have had the benefit of previous developmental exercises and subsystem qualification. Field hangar assembly and checkout and mating on the pad is shown extended to two or three days prior to launch. During this period, the nominal launch date to within several days would be selected. By the end of this period, the actual launch date and time would be known (barring launch holds). Also at the end of this period, with the start of

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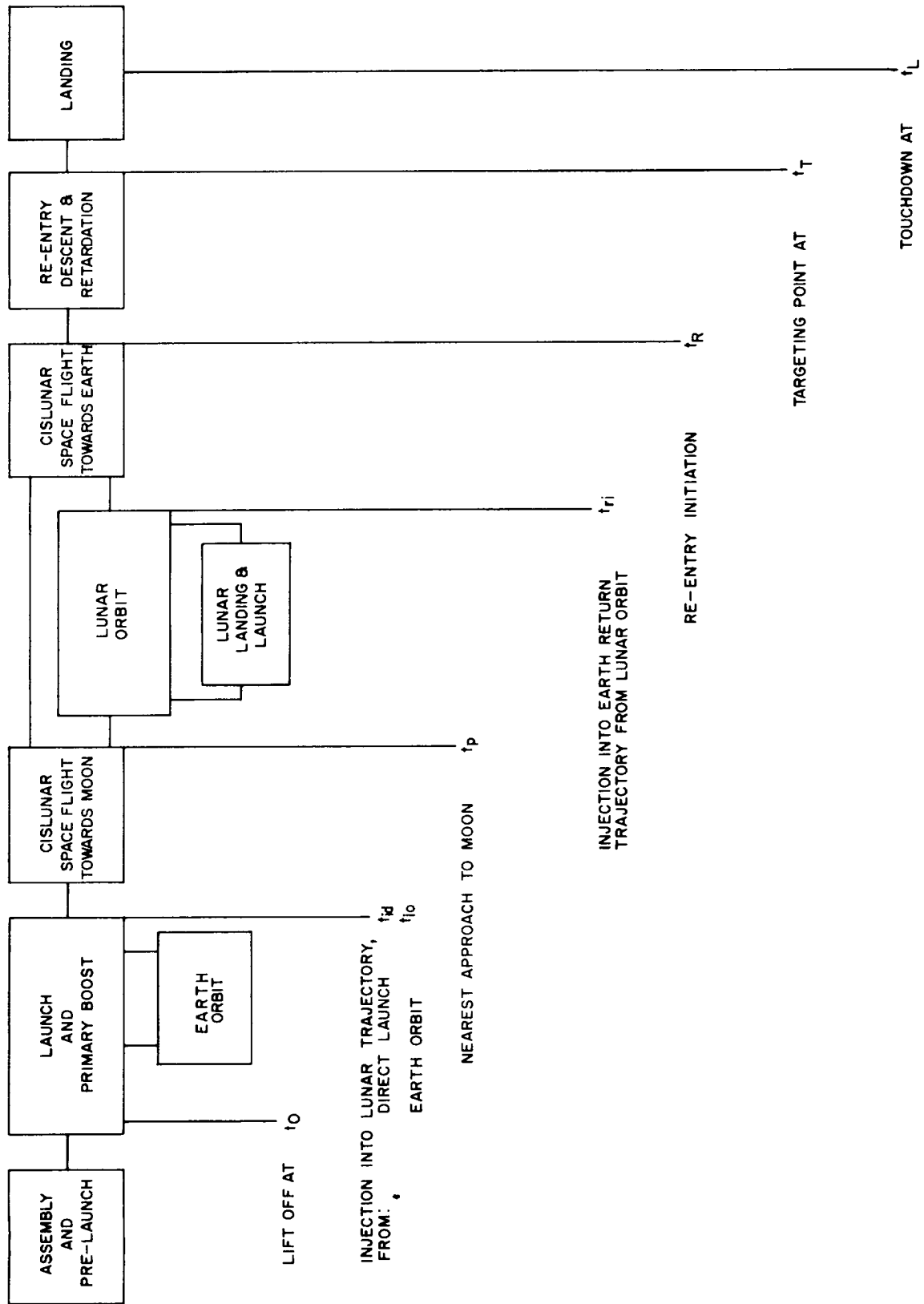


Figure I-1-1. Summary of operational APOLLO missions

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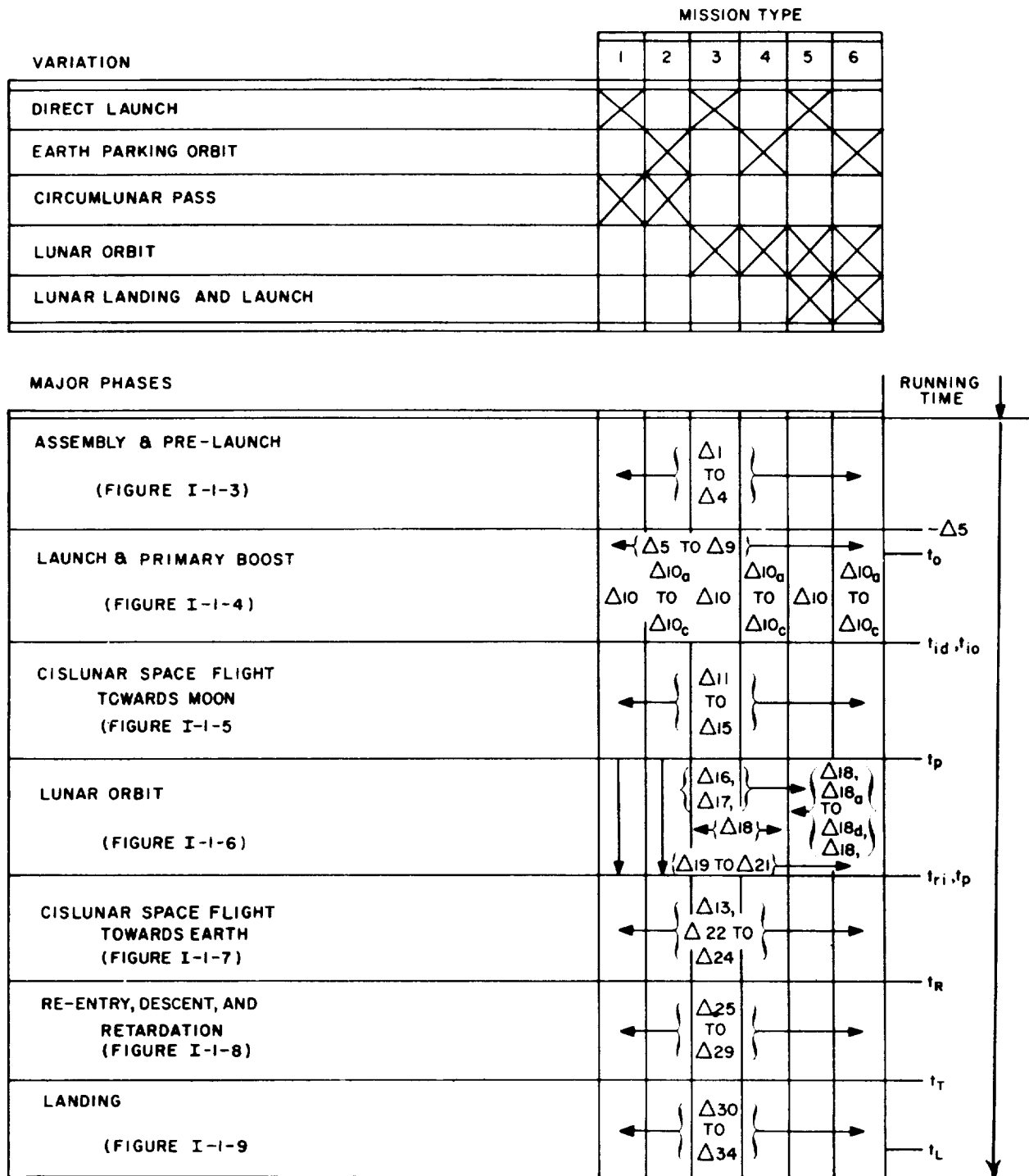


Figure I-1-2. Mission variation and phase timing summary

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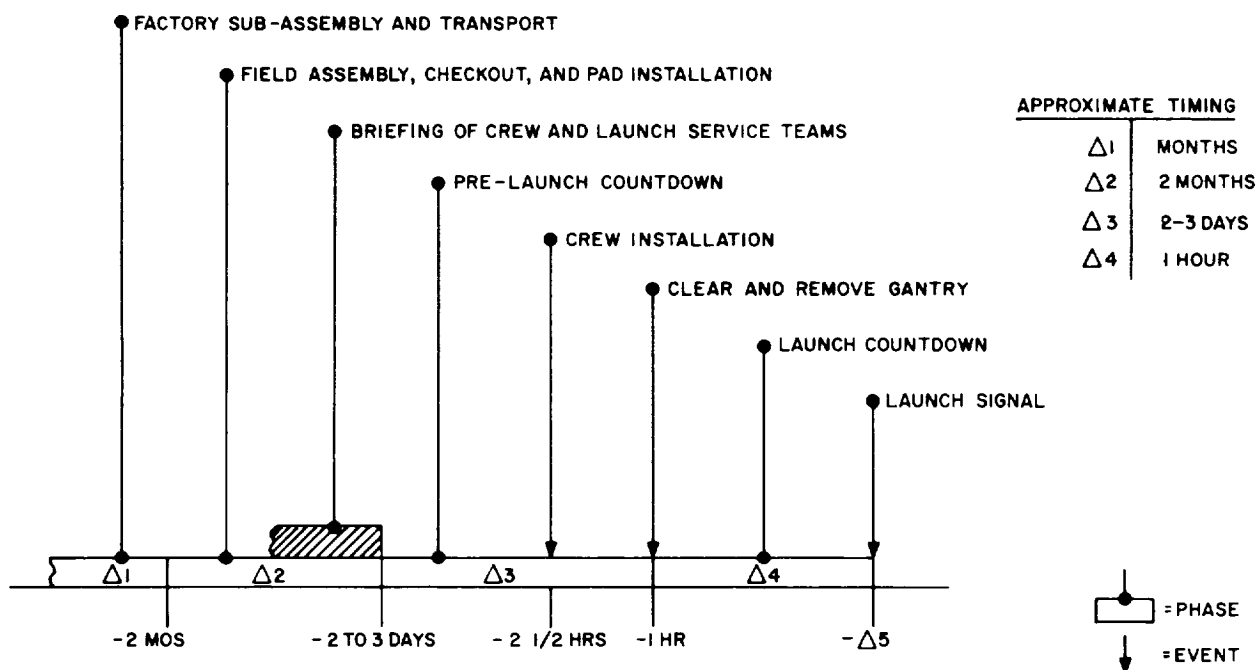


Figure I-1-3. Phase definition—assembly and pre-launch

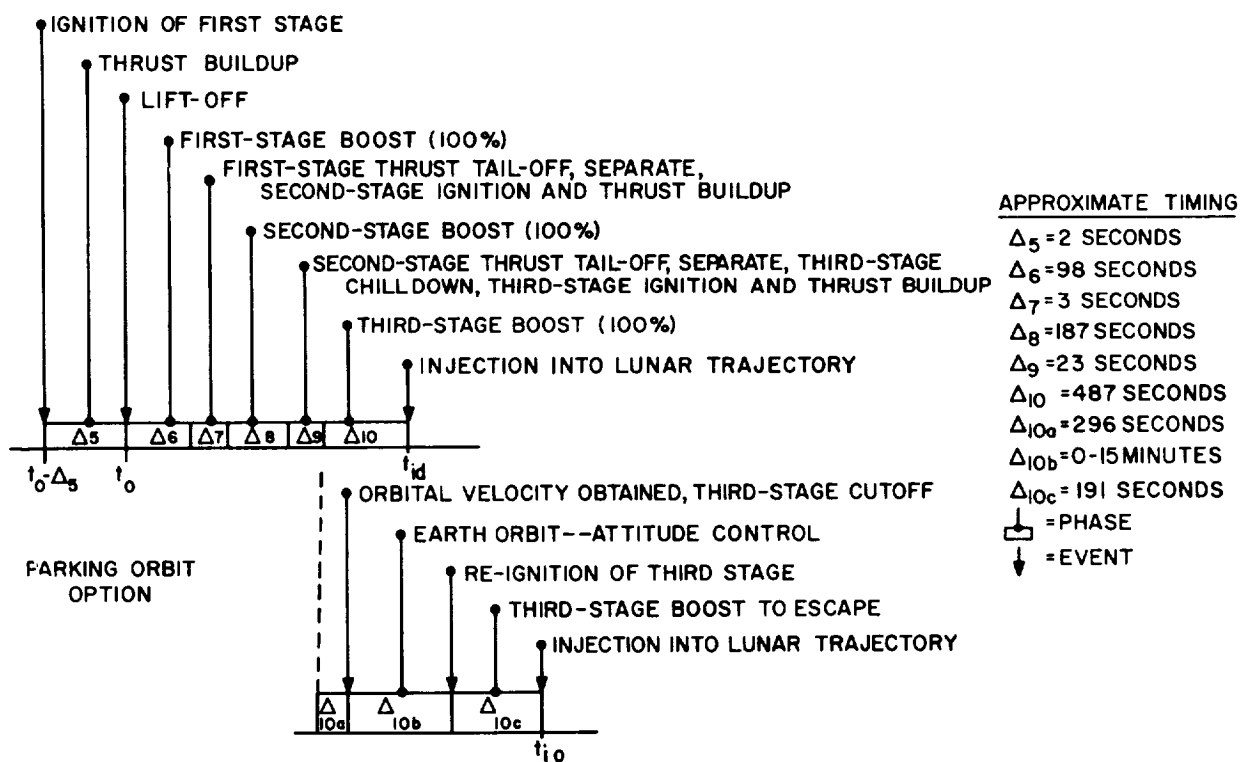


Figure I-1-4. Phase definition—launch and primary boost

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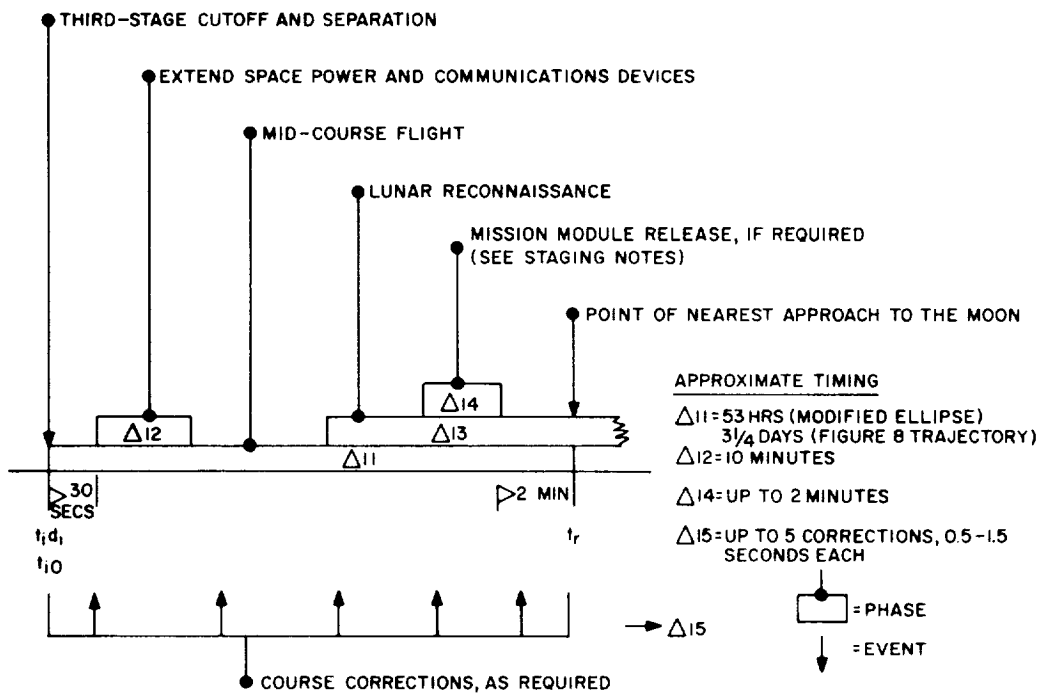


Figure I-1-5. Phase definition—cislunar flight towards Moon

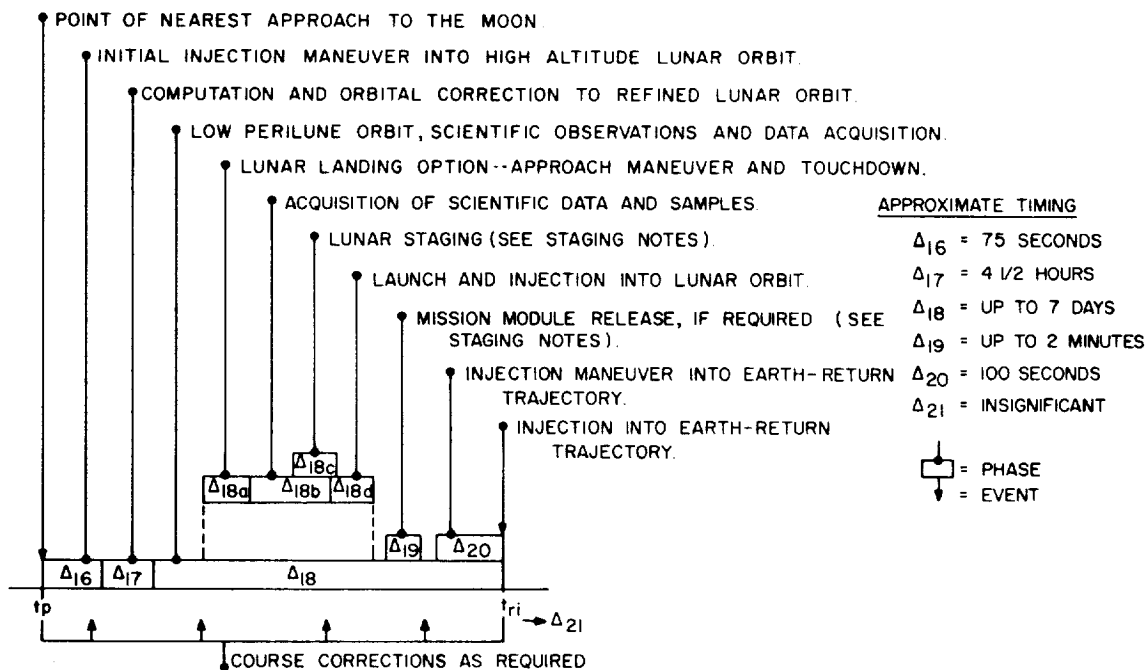


Figure I-1-6. Phase definition—lunar orbit

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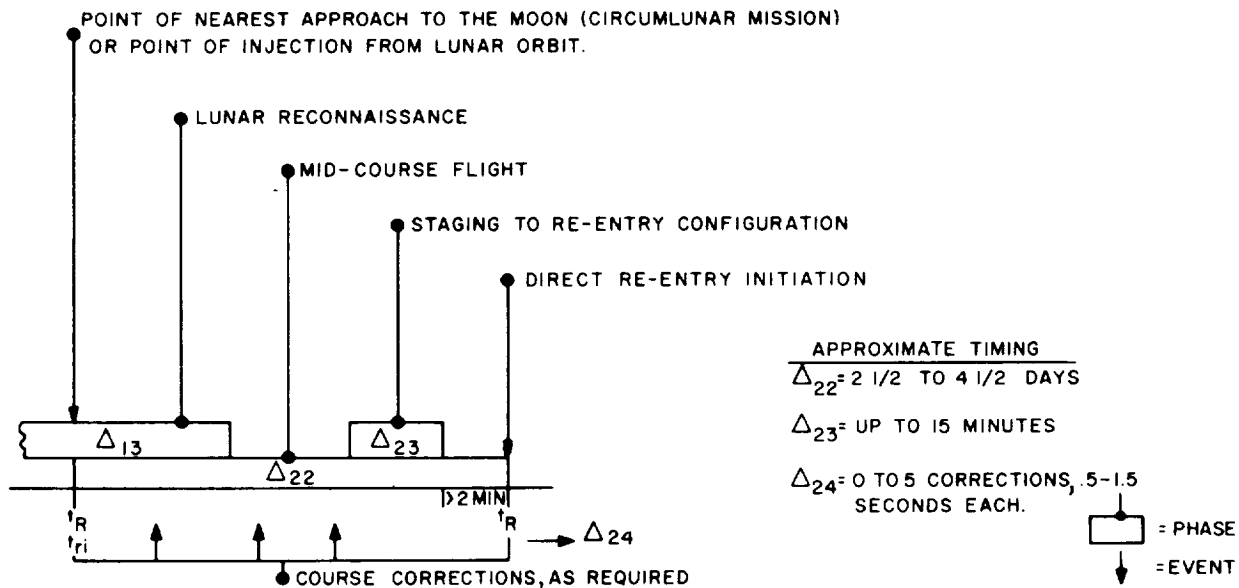


Figure I-1-7. Phase definition—cislunar space flight towards Earth

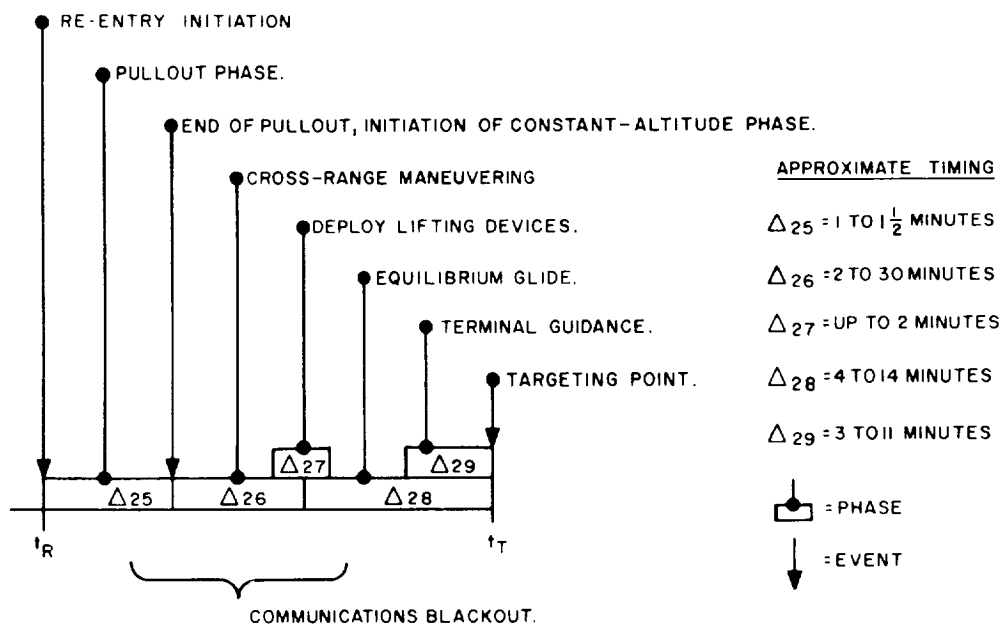


Figure I-1-8. Phase definition—re-entry, descent and retardation

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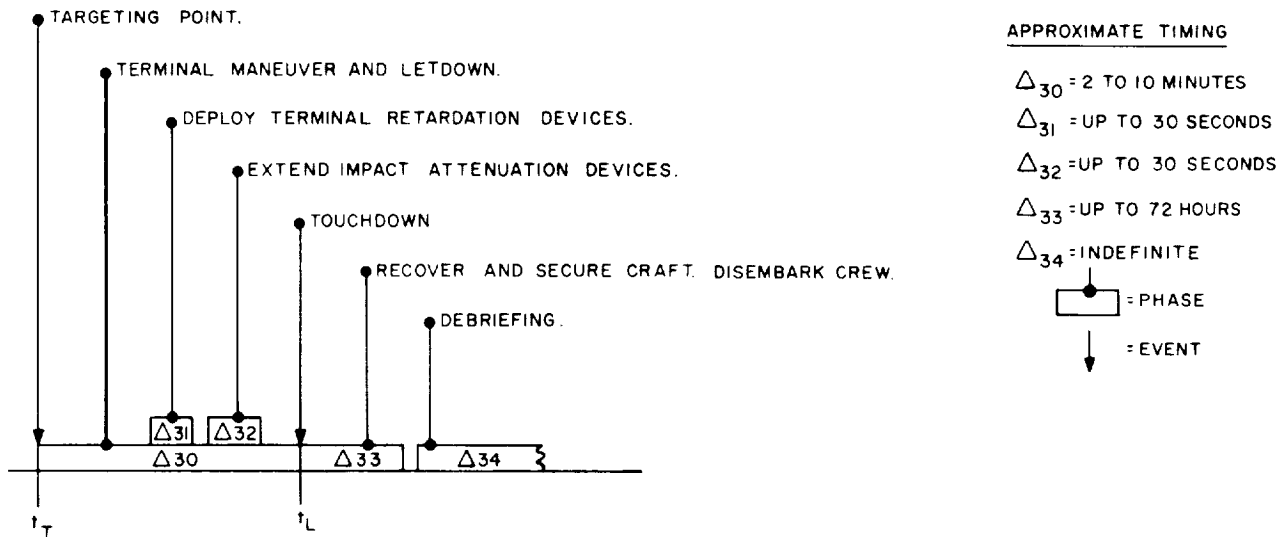


Figure I-1-9. Phase definition—landing

pre-launch countdown, crew training and briefing would effectively end, except for last-minute details, in order that they may participate in the countdown procedures. Pre-launch countdown will proceed to about one hour before launch. During this phase, the launch service crew will conduct a complete system and detailed subsystem checkout. The crew would be installed at about 2-1/2 hours before launch to assist in the checkout, and check cabin sealing. The launch countdown phase will be characterized by fueling, equipment turn-on, and disconnect activities up to the time of launch.

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The launch and primary boost regime (Figure I-1-4) is characterized by the parking orbit option prior to injection into the lunar trajectory. Actual lift-off at t_0 will occur after booster thrust has been built up to near 100 percent. The thrust buildup time for the Saturn booster will be approximately two seconds. This period will be especially critical from the safety standpoint. After a first-stage boosting period, of about 98 seconds, the first-stage thrust must be allowed to tail off before the second stage is ignited. The total transfer time from first to second-stage boost will be approximately three seconds. After a second-stage boosting period of about 187 seconds a transfer to the third stage will take place in about 23 seconds. The greater part (≈ 20 seconds) of this time will be required for chilldown of the third stage. In a typical direct launch, injection will occur after 487 seconds of third-stage boost. Injection into a typical parking orbit could be made after 296 seconds of third-stage boost. Coasting in the earth orbit would be limited to about 15 minutes maximum before a final boost of the third stage for about 190 seconds to escape. During the transfer time between stages, successive abort rocketry will be released (see Appendix SC-A).

Cislunar flight towards the Moon will be defined as passage from the point of injection after a direct or earth orbit launch to a point of nearest approach to the Moon (Figure I-1-5). This point will vary depending on the subsequent mission and will be approximately 1000-2000 miles from the lunar surface. The time duration of the flight ($\Delta 11$) will vary from about 53 hours for a modified ellipse trajectory to about 3 and 3/4 days for a "figure-eight" trajectory. Space power and communications devices will be extended as soon as possible; however, a minimum delay is required before extension to provide for adequate separation from the primary booster. Extension time ($\Delta 12$) up to perhaps 10 minutes may be desirable in order to minimize drive power requirements. Lunar reconnaissance can begin as soon as meaningful visual sightings are possible and within practical electrical power limits where active devices are used. Mission module release could conceivably be a requirement. (These are treated in Appendix SC-A.) If such is the case, the release should be accomplished before about two minutes of reaching the point of nearest approach to the Moon where a lunar orbit is to be obtained. This time will be required for vehicle orientation to the correct maneuver attitude. The time for mission module release might extend to as much as two minutes in order to obtain an attitude at separation which could be used (with appropriate retro boost) to inject the module into a long-term, useful trajectory. Course corrections will be required as a consequence of deviations from the nominal trajectory. There may be as many as five corrections, at 1 g for about 0.5 to 1.5 seconds each.

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The lunar orbit mission with a possible future option for a lunar landing are expressed in Figure I-1-6. For a lunar orbit, a maneuver into an initial orbit must be made. The total velocity increment required to obtain a refined orbit (50 nmi perilune altitude, 1000 nmi apolune altitude) will require on the order of 100 seconds of 1 g thrust. With no theoretical fuel penalty, this could conceivably be done incrementally over two or more steps. The model selected here assumes a single correction extending over approximately 75 seconds ($\Delta 16$) to give a near circular orbit at 1000 nmi altitude, and a single orbit over $\Delta 17$ prior to correction into the refined orbit. The refined orbit time ($\Delta 18$) of up to 7 days will allow up to 50 orbits. Perilune will occur over the near side of the Moon, in sunlight, in order to obtain maximum reconnaissance capability. It is further assumed in this model that the injection into the earth-return trajectory is done during one thrusting period ($\Delta 20$). Obviously a large number of variations to this procedure prevail which could significantly alter the time apportionment between $\Delta 17$, $\Delta 18$, and $\Delta 20$; further, by including more incremental orbits in the progression into and out of the refined orbit, more phases than shown can be generated. Release of mission module over a time period up to 2 minutes ($\Delta 19$) is a possible requirement for the same reasons given in the discussion of the cislunar flight. In this case, the mission module would be best left in orbit with appropriate instrumentation and telemetry. Course corrections throughout the lunar orbit period would be necessary for proper maintenance of orbit. They would each be quite small compared to the maneuvers into and out of orbit.

The lunar landing option is indicated in Figure I-1-6 for reference, but no attempt will be made at this time to assign values to the time intervals. The transfer into a lunar landing is assumed to follow from a refined lunar orbit and a lunar launch to result in injection back into the refined orbit before return. The total time for $\Delta 18a$, $\Delta 18b$, and $\Delta 18c$ would be limited to around 5 days in an over-all mission length of 14 days.

The phasing of the return flight to earth is shown in Figure I-1-7, starting from the point of closest approach to the Moon, at time t_p , for the circumlunar mission or at injection from the lunar orbit, at time t_{ri} . Lunar reconnaissance will have continued as before, terminating when no longer feasible from an electrical power or visibility/resolution standpoint. The mid-course return flight time to point of re-entry initiation will require about 2-1/2 to 4-1/2 days. During the return, data transmission will be required for the records obtained, but not transmitted previously, either because of far-side blackout or slow-time transmission for reduced bandwidth. Staging for re-entry will be required just prior to re-entry with enough time (≈ 2 minutes) for reorientation of the heat shield into

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the flight path. Staging to the re-entry configuration could conceivably require a period of 15 minutes or more to jettison the mission module, propulsion, and fairing, depending on the total configuration and a possible tradeoff between automatic-release-devices weight and requirements for last-minute course corrections. Course corrections prior to staging for re-entry will probably be required; the requirement may be estimated at upwards of 5 corrections at 1 g for about 0.5 to 1.5 seconds each.

Timing during re-entry, descent, and retardation will be a direct function of vehicle configuration. The times given for $\Delta 25$, $\Delta 26$, and $\Delta 28$ in Figure I-1-8 are representative over the range of L/D from 0.25 to 0.75. For certain re-entry modes, the non-maneuvering pull-out phase will terminate when altitude rate first becomes zero. During the constant altitude phase of such a mode, limited maneuverability is possible. This phase will continue until equilibrium glide conditions are established in terms of altitude, velocity, and wing loading. Around the time of transition into equilibrium glide, lifting devices can be deployed. Reasonably, the extension time for the lifting devices could be as long as two minutes in view of the heavy prevailing loads. The equilibrium glide has been carried to a targeting point. This point is defined here as the objective from which the terminal maneuver to a prepared landing site can be made. It will lie in the range of altitudes between about 50,000-100,000 feet, depending on the landing characteristics of the vehicle.

Extending over the constant-altitude phase and over parts of the pullout and equilibrium glide phases is a region of communications blackout. The extent and location of this region will be a function of frequency and configuration. After passing through this region, communication contact will allow terminal guidance aid through ground assist. Nominally, self-contained guidance would be used up to the targeting point, and terminal guidance thereafter. However, a transition into the terminal condition is desirable as soon as possible. Conceivably, the terminal guidance capability can be available up to short times before reaching the targeting point.

The sequence of events and phases during landing (Figure I-1-9), have been generalized to account for vertical or glide landings. Total time for the terminal maneuver to touchdown will range from about 2 to 10 minutes, depending on configuration and form of retardation. During this time, retardation devices such as parachutes will be deployed from the low L/D vehicles. These have a minimum time for extension, reasonably within 30 seconds. Impact attenuation devices will, likewise, have such a limiting time. At touchdown, recovery services immediately secure the craft and disembark the crew in the

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event of a planned site landing. For the non-nominal landing, search time and recovery team deployment time could result in a three-day delay before recovery. Debriefing of crew and service teams would commence as soon as practical.

1.2 FUNCTIONAL STATUS AND OPERATION

During the APOLLO mission, there will be a number of required functions which must be implemented by the crew, vehicle subsystems, and a ground complex. The control of the mission, by phase, will be based on operating programs which will govern the activity of each functional element, including command priority assignments.

A generalized catalogue of nominal mission operating programs is presented by phase, in Table I-1-I. The phases correspond, by number, with those defined in the previous section by Δ subscripts. The assignment of activities and priorities at this time is preliminary, of course. The intent here is to develop an adequate perspective for an integrated preliminary design. It should be noted that, just as the phases overlap in time (as expressed in Figures I-1-3 through I-1-9, inclusive), so too will the operating programs.

TABLE I-1-I. MISSION OPERATING PROGRAMS

(Only the more significant programs are indicated for each phase.
In most cases, these programs would be crew initiated.)

| | | |
|----|--|--|
| 1. | Factory sub-assembly and transport. | Production control and testing procedures. Transportation program. Training program. |
| 2. | Field assembly checkout and pad installation. | Checkout and installation procedures. Ground support maintenance and test programs. |
| 3. | Pre-launch countdown. | Pre-launch countdown procedures. Search and recovery forces deployment. |
| 4. | Launch countdown. | Detailed launch countdown procedures, including launch hold checkpoints. |
| 5. | Thrust buildup. | Automatic booster program with abort criteria checks in vehicle and on ground. |
| 6. | First-stage boost (100 percent). | Primary booster program with abort criteria checks in vehicle and on ground. |
| 7. | First-stage thrust tail-off, separate, second-stage ignition and thrust buildup. | |
| 8. | Second-stage boost (100 percent). | |

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TABLE I-1-I. MISSION OPERATING PROGRAMS (Continued)

| | | |
|-----|--|---|
| 9. | Second-stage thrust tail-off, separate, third-stage chill-down, third-stage ignition and thrust buildup. | } Primary booster program with abort criteria checks in vehicle and on ground. |
| 10. | Third-stage boost (100 percent). | |
| 10a | Orbital velocity obtained, third-stage cutoff. | Alternative primary booster program with abort criteria checks in vehicle and on ground. |
| 10b | Earth orbit attitude control. | Automatic attitude hold program (vehicle), with crew assist. Quick checks for nominal mission continuation. |
| 10c | Third-stage boost to escape. | Alternative primary booster program with abort criteria checks in vehicle and on ground. |
| 11. | Midcourse flight. | Space flight programs - crew initiated and monitored. Automatic and manual performance. |
| 12. | Extend space power and communications devices. | Manual, through electromechanical linkage. |
| 13. | Lunar reconnaissance. | Manual procedures. |
| 14. | Mission module release, if required. | Manual operation - attitude control and release mechanisms. |
| 15. | Course corrections, as required. | } Manually initiated, automatic control. |
| 16. | Initial injection maneuver into high altitude lunar orbit. | |
| 17. | Computation and correction to refined lunar orbit. | Short-term space flight program. Manual initiation and automatic control of correction. |
| 18. | Low perilune orbit scientific observations and data acquisition. | Space flight programs, primary manual. |
| 19. | Mission module release, if required. | Manual operation - attitude control and release mechanism. |
| 20. | Injection maneuver into earth-return trajectory. | } Manually initiated, automatic control. |
| 21. | Course corrections as required. | |
| 22. | Midcourse flight. | Space flight programs-crew initiated and monitored. Automatic and manual performance. |
| 23. | Staging to re-entry configuration. | Manual operation-attitude control and release mechanisms. |
| 24. | Course corrections, as required. | Manually initiated, automatic control. |

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TABLE I-1-I. MISSION OPERATING PROGRAMS (Continued)

| | | |
|-----|---|---|
| 25. | Pullout Phase. | Automatic attitude control. Search and recovery force deployment. |
| 26. | Cross-range maneuvering. | Semi-automatic guidance. |
| 27. | Deploy lifting devices. | Manual operation, emergency automatic takeover. |
| 28. | Equilibrium glide. | Semi-automatic guidance. |
| 29. | Terminal guidance. | Semi-automatic guidance with ground assist. |
| 30. | Terminal maneuver and let-down. | Manual operation; with ground control for near-nominal landing. |
| 31. | Deploy terminal retardation devices. | Manual operation, emergency automatic takeover. |
| 32. | Extend impact attenuation devices. | |
| 33. | Recover and secure craft. Disembark crew. | Search program. Recovery tactical plan. |

Definition of the functions of the crew and elements of the vehicle and ground complex is given in Appendix SC-B. These definitions are made intentionally broad in order to avoid unnecessary restrictions on subsystem design. It should be noted that the names assigned to the functional elements are generic and do not have a one-to-one correlation with an implementing subsystem in all cases.

The status and operation of the functional elements are presented, by phase in Table I-1-II. Use of this table for any one-type mission would naturally follow the progression indicated in Figure I-1-2. Functional status and operations are given only for the more significant or unique aspects relative to each phase. The general or ordinary operations can be surmised from the definitions in Appendix SC-B (e. g. intra-crew communications, computation, maintenance).

1.3 ENVIRONMENT

The external environments to which it is anticipated the APOLLO vehicle will be exposed during the lunar orbit mission are presented here in accordance with the various phases comprising the mission profile, from factory through recovery. This profile not only serves as a guide to preliminary design but also provides a basis for environmental design requirements.*

*The environmental design requirements for the vehicle and subsystems are specified according to the direct exposure environment. The latter will differ from the mission environment by virtue of (a) attenuation due to shipping containers, shockmounts, radiation shielding, and the like; and, (b) mutual effects such as radio noise, vehicle-atmosphere contamination, vehicle environmental control, etc.

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TABLE I-1-II. FUNCTIONAL STATUS AND OPERATION

| FUNCTIONAL STATUS AND OPERATIONS | | | | | | | | | | | | |
|---|-------------------------|--|---|------------------------------------|-------------------------|--|------------------|------------------------|----------------------|--|--|---|
| Mission Phase | Time Duration (Approx.) | Running Time (From Lift-off to Start of Phase) | MAJOR VEHICLE SUBSYSTEMS | | | | | GROUND SUPPORT COMPLEX | | | | |
| | | | Crew | Instrumentation and Communications | Navigation and Guidance | Flight Control | Electrical Power | Crew Services | Landing and Recovery | Launch and Landing Service | Communications Tracking, Search, and Retrieval | |
| 1. Factory sub-assembly and transport | Months | Months | Training and observation | → | → | Not applicable | → | → | → | Training | → | → |
| 2. Field assembly, checkout and pad installation | 2 Months | 2 Months | Briefing and observation | → | → | Final assembly, checkout, and installation | → | → | → | Training. On-serve assembly and checkout. Install vehicle on pad. Landing service in training | → | → |
| 3. Pre-launch countdown | 2-3 Days | 2 to 3 Days | Board (at 2 1/2 hrs.). Participate in pre-flight checkout | → | → | Calibration, adjustment, and pre-flight checkout | → | → | → | Pre-flight checkout and sealing | → | → |
| 4. Launch countdown | 1 Hour | 1 Hour | Participate in final subsystem checks. Monitor countdown. Prepare for boost | → | → | Final checks and procedures | → | → | → | From - 24 hrs. - Final check of re-motor forces, monitor status up to launch. Deploy launch vehicle to search and recovery centers | → | → |
| 5. Thrust buildup | 2 Seconds | 2 Seconds | Monitor performance of booster and subsystems | → | → | Alignment and time synchronization | → | → | → | Transfer to maximum restraint | → | → |
| 6. First stage boost (100%) | 98 Seconds | 0 Seconds | Monitor performance of booster and subsystems | → | → | Running time monitor | → | → | → | Auxiliary vehicle power | → | → |
| 7. First stage thrust tail-off, separate, second stage ignition and thrust buildup | 3 Seconds | + 98 Seconds | Monitor separation and second stage ignition | → | → | Backup computation of booster guidance | → | → | → | Conclude launch services | → | → |
| 8. Second stage boost (100%) | 187 Seconds | + 101 Seconds | Monitor performance of booster and subsystems | → | → | Timing checks | → | → | → | Landing service on alert | → | → |
| 9. Second stage thrust tail-off, separate, third stage child-off, ignition and thrust buildup | 23 Seconds | + 288 Seconds | Monitor separation and third stage ignition | → | → | Backup computation of booster guidance | → | → | → | → | → | → |
| 10. Third stage boost (100%) | 487 Seconds | + 311 Seconds | Monitor performance of booster and subsystems | → | → | Backup computation of booster guidance | → | → | → | → | → | → |
| 10a. Orbital velocity obtained, third stage cutoff | 294 Seconds | + 311 Seconds | | → | → | → | → | → | → | → | → | → |
| 10b. Earth orbit - attitude control | 0-15 Minutes | + 607 Seconds | | → | → | Running time monitoring and attitude control and backup attitude control | → | → | → | → | → | → |
| 10c. Third stage boost to escape | 191 Seconds | + 10 to + 25 Minutes | | → | → | Backup computation of booster guidance | → | → | → | → | → | → |

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TABLE I-1-II. FUNCTIONAL STATUS AND OPERATION (Continued)

| | | | | | | | | | | | | |
|--|--|--|--|---|---|---|---|---|--|-----------------------------|--|---|
| 11. Mid-course flight | 53 Hours (Modified ellipse), 3-1 4 Days (Figure 6) | At t_0 or t_p $t_0 = 798$ Seconds $t_p = 13$ to 28 Minutes | Scientific observations, monitoring and recording. Science and record maintenance. Navigation and computation assist. | Acquisition of scientific data, recording, and transmission. Crew and subsystems. Interim black-out due to vehicle shadowing at far side of Moon. | Eggers' determination and guidance computation (see Phase 15). Continuity of view. | Attitude control (semi-automatic or manual) for solar collector, observational aiming. | Auxiliary power unit after Phase 12. Main power for critical functions during Moon shadowing. Battery charging. | Transfer to "shirt-sleeves" environment after eating, exercise, waste disposal, radiation protection. | Turned off immediately. | Landing service on standby. | Monitor and analyze crew and subsystem performance. Assume command and control. Provide redundant navigational data. Interim black-out due to vehicle shadowing. | Blackout when vehicle is behind Moon. |
| 12. Extend space power and communication devices | 10 Minutes | Minimum of t_0 $t_0 = 30$ Seconds | Control of operation. | Possible frequency switch. Aid in antenna aiming. Space station. | For vehicle-limited devices, Sun or Earth seeking attitude. | Attitude control (manual or semi-automatic) lock-on. | Transfer to main vehicle power. | Not applicable. | | | Possible frequency switch. Aid in Earth-based antenna lock-on. | Not specifically applicable. |
| 13. Lunar reconnaissance | Minimum of 2 Days | Between t_0 and t_p | Scientific observation, recording, report. Attitude control. | Acquire, record, and transmit scientific data. | Not generally applicable. Attitude control assist. | Attitude control (manual or semi-automatic). | Main vehicle power, battery over to auxiliary during vehicle or Moon shadowing. | | | | Receive data record. | |
| 14. Mission module release, if required. | Up to 2 Minutes | Maximum of t_0 - 4 Minutes | Perform separation procedures. | Command data for release. Status reporting on operation. | Prior computation for time and attitude at release. Set mission module on desirable trajectory. | | Auxiliary power for all critical functions during release. | Cut back on release facilities. Modification in waste disposal during release facilities. | | | Prior computation and command data to vehicle on release aiming. | |
| 15. Course corrections, as required | Up to 5 corrections 75 Seconds | Between t_0 and t_p $t_p = 2.3$ to 3.3 days | Prior assist to navigation. Monitor automatic corrections. | Prior redundant navigational data from the ground. | Computation and control of corrections. | Thrusting as required. Attitude hold. | Auxiliary power for all critical functions during corrections. | Temporary partial restraint. | | | Prior computation and transmission data for computation and transmission of navigational data. | |
| 16. Initial injection maneuver into high altitude lunar orbit. | Up to 75 Seconds | At t_0 $t_0 = 2.3$ to 3.3 days | Prior assist to navigation and computation. Monitor automatic maneuver. | Prior computation of 3Y record of maneuver. | Thrusting and attitude hold. | Auxiliary power for all critical functions during maneuver. | | | | | | |
| 17. Computation and orbital correction to refined lunar orbit. | 4-1 2 Hours | t_p - 75 Seconds | Scientific observation, recording, computation. Monitor automatic maneuver. | Intermittent capsule. Limited acquisition of data, record. | Navigation, computation for correction to refined orbit. | Attitude control (manual or semi-automatic). Thrusting during correction over to auxiliary corrections. | Main vehicle power, battery over to auxiliary during maneuver. | Shirt-sleeves environment. Partial restraint during corrections. | | | Reduced capability for critical functions (ground noise and intermittent total blackout (far side)). | Intermittent capsule in vicinity of Moon. |
| 18. Low Perilune orbit, scientific observations and acquisition. | Up to 7 Days | t_p - 4-1 2 Hours | Scientific observation, recording, report. Attitude control. Flight operations, subsystem monitoring and over-ride. Maintenance and repair by astronaut. | Acquire, record scientific data. Transmit data record. Status reporting (Vehicle-Moon). | Orbit and attitude computation. | Attitude control (manual or semi-automatic). | | Shirt-sleeves environment. | Check-out and put on standby during scheduled. | | | |
| 19. Mission module release, if required. | Up to 2 Minutes | t_p - (1 2 to 7 Days) | Perform separation procedures. | Not specifically applicable. | Prior computation for time and attitude at release. Set mission module trajectory. | | Auxiliary power for all critical functions during release. | Cut back on release facilities. Modification in waste disposal and aiming facilities. | | | | |

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TABLE I-1-II. FUNCTIONAL STATUS AND OPERATION (Continued)

| Mission Phase | Time Duration (approx.) | Crew | MAJOR VEHICLE SUBSYSTEMS | | | | | GROUND SUPPORT COMPLEX | | | |
|--|-------------------------|---|---|--|--|---|---|--|--|----------------------------|--|
| | | | Instrumentation Communications | Navigation Guidance | Flight Control | Electrical Power | Crew Services | Landing Recovery | Launch and Landing Service | Communications Telemetry | Tracking, Search, and Retrieval |
| 20. Injection maneuver into Earth return trajectory. | 100 Seconds | Prior assist to crew (12 to 15) Days | Prior assist to crew (12 to 15) Days | Prior computation of 3V required. Control of maneuver | Thrusting and attitude hold | Auxiliary power to all critical functions during maneuver | Temporary restraint | | | | |
| 21. Course corrections, as required. | Insignificant | Prior assist to crew (12 to 15) Days | Prior recomputation of 3V required. Control of corrections | Prior computation of 3V required. Control of corrections | Thrusting and attitude hold | Auxiliary power to all critical functions during corrections | Temporary restraint | | | | |
| 22. Mid-course flight | 2.1 to 4.2 Days | Prior assist to crew (12 to 15) Days | Acquisition of scientific data and guidance record. Transmission of accumulated data record. Att-hold on command by crew. | Epimeric drive. Mission and guidance computation (see Phase 24). Att-hold on command by crew. | Attitude control (semi-automatic or manual) for solar collector, antenna, or observation. Att-hold on command by crew. | Main power with intermittent switchover to auxiliary during Phase 24 and after Phase 23 | "Shir" services" (entire back after reentry staging) | Checked and put on standby just prior to reentry. reentry. | Monitor and assist crew and subsystem performance. As- data. Normal trajectory monitoring. Impact prediction. Start deployment of search and recovery teams just prior to reentry. | | |
| 23. Staging to reentry configuration. | Up to 15 Minutes | Perform staging procedures. Att-hold on command by crew (12 to 15) Days | Status reporting on operation timing and attitude. Post-aid frequency switch. | Prior computation of time and attitude. Mission module on desirable trajectory. | Attitude control (manual or semi-automatic) | Transfer to auxiliary power | Cut back on facilities to maximum restraint for reentry | Not applicable | Not specifically applicable | Possible frequency switch. | Not specifically applicable. |
| 24. Course corrections, as required. | Up to 5 seconds | Prior assist to crew (12 to 15) Days | Prior recomputation of 3V required. Control of corrections | Computation and control of corrections | Thrusting, as required. Attitude hold. | Auxiliary power for all critical functions during corrections | Temporary partial restraint | | Prior computation and transmission of redundant navigational data | | Prior tracking - data for computation of redundant navigational data |
| 25. Pollout. | 1.1 to 2.2 Minutes | Limited capability (12 to 14) Days | Support navigation and communications until blackout. Possible frequency switch. | Computation, control of angle-of-attack for reentry corridor. Limited range and cross-range control. | Attitude control (automatic). Reaction jets. | Auxiliary power Maximum restraint | On standby | Landing service on alert | Support navigation and guidance-impact prediction. Search and recovery forces. Impact prediction. Computation for impact prediction | | Tracking data for impact prediction |
| 26. Cross-range maneuvering. | 2.30 Minutes | Prior assist to crew (12 to 15) Days | Limited capability. | Computation and control of guidance data. | Attitude and flight path control (semi-automatic). Display of guidance characteristics | | | | | | Computation for impact prediction. |
| 27. Deploy lifting devices. | Up to 2 Minutes | Prior assist to crew (12 to 15) Days | Facility for indicating conditions for initiation. | | Deploy action | | | | | | Not specifically applicable |
| 28. Equilibrium glide. | 4 to 14 Minutes | Prior assist to crew (12 to 15) Days | Establish terminal guidance contact as soon as possible | Guide to targeting point. Transition from self-contained to terminal guidance | Coordinated reaction and aerodynamic controls | | | | Computation for impact prediction and terminal guidance | | Tracking in prediction and terminal guidance. Search and recovery forces |
| 29. Terminal guidance. | 3 to 11 Minutes | Prior assist to crew (12 to 15) Days | Provide reliable terminal guidance based on best estimate of redundant data. | Range and cross-range parameter display. | Transition into terminal guidance primary method | | | | | | |
| 30. Terminal maneuver and lockdown. | 2 to 10 Minutes | Prior assist to crew (12 to 15) Days | Perform maneuver in accordance with ground control information (nominal, or by visual sightings from nominal). | Provide maneuver flight control. Signal for deployment of retardation devices. (Phase 31) | Lifting surfaces | | | Activate beacon and other in-flight location devices. Also, Phase 31 and 32. | Maneuver commands for landing site. Field devices | | |

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TABLE I-I-II. FUNCTIONAL STATUS AND OPERATION (Continued)

| | | | | | | | | | | | | | | |
|---|------------------|--|---|--|---|---|--|--|--|--|---------------------------------------|---|---|---|
| 31. Deploy terminal retardation devices. | Up to 30 seconds | Between t_T and t_L | Perform operation | Not specifically applicable | Attitude control (stabilization) | Attitude control (automatic) | | | | | Terminal retardation devices deployed | Not specifically applicable | Not specifically applicable | Not specifically applicable |
| 32. Extend impact attenuation devices. | Up to 30 seconds | Between t_T and t_L | | | | | | | | | Impact attenuation devices extended | | | |
| 33. Recover and secure craft. Dismantle crew. | Up to 72 hours | At t_L $t_L = 19 \pm 1$ (10 to 60) Minutes | Secure vehicle, shut down all unnecessary systems, aid in search and recovery. Dismantle. | In support of search and recovery operations | Equipment of possible assist in location efforts. | Not applicable | | | | | Survival unit | Recovery at aid in effect, prepared site. | Aid in location | Search completed. Recovery of: 1. Crew 2. Data 3. Vehicle, in order. |
| 34. Debriefing | Indefinite | t_L | Psycho-medical analysis, interviews and reports. | | | Disassembly, analysis of equipment condition. Analysis of performance from vehicle records. Report. | | | | | | | Analysis of mission performance record. Report. | |

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From a virtually unlimited number of environmental factors, a selection can be made of those significant to the particular mission. Significance of a factor is closely tied to the location of the vehicle and does not extend over the complete mission. A summary of such significance, by phase, is shown in Table I-1-III. The squares left blank denote cases for which the listed factor is not applicable or is negligible (in magnitude or possible effect). Among the space environmental factors which do not appear significant to the APOLLO mission are asteroids, comets, interplanetary dust clouds, solar wind, the ionosphere, and ozone.

A detailed summary of the environment, by phase, is presented in Table I-1-IV. The factors and appropriate summary references of Table I-1-IV are discussed briefly in the following paragraphs.

1.3.1 Acceleration and Shock

Acceleration profile data is based on Saturn data, a nominal range of on-board propulsion capability, and acceptable limits for preserving the functional capability of the crew during re-entry. Shock data includes considerations of ground handling procedures and estimates for on-board propulsion rise-time, retardation and impact.

1.3.2 Vibration and Acoustic Noise

These factors are important from the standpoint of human tolerance and material fatigue. Vibration estimates are based on Category B transport by common carrier (Reference 1) and extrapolated data from other boosters. Transonic instabilities during re-entry are a potential problem. Acoustic noise is limited to the atmosphere and will be relatively significant only during powered boost and re-entry. Noise contribution is by the booster engine and boundary layer turbulence. Maximum values are estimated from Saturn data, experience with other boosters, and information contained in references 2 and 3.

1.3.3 Pressure and Density

Pressure during phase 1 is taken for conditions of transportation to the launch site, and during phases 2 through 5 for a reasonable variation at Cape Canaveral. Density on the ground does not appear to be especially applicable. In-flight, the ARDC Model Atmosphere, 1959 is used with an extension out to 20,000 nmi (see Figures I-1-10 and I-1-11 taken from Reference 4). Beyond 20,000 nmi both pressure and density would certainly appear negligible.

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TABLE I-1-III. SIGNIFICANCE OF ENVIRONMENTAL FACTORS BY PHASE

| Environmental Factor | Phase | Acceleration | Shock | Vibration | Acoustic Noise | Pressure | Density | Temperature | Gravity | Ionizing Radiation | Meteoroids | Geomagnetic Field | Local Winds | Salt Spray | Sand & Dust | Precipitation | Humidity |
|--------------------------|-------|--------------|-------|-----------|----------------|----------|---------|-------------|---------|--------------------|------------|-------------------|-------------|------------|-------------|---------------|----------|
| | | | | | | | | | | | | | | | | | |
| | 1 | | X | X | | X | | X | | | | | | X | X | X | X |
| | 2 | | X | X | | X | | X | | | | | | X | X | X | X |
| | 3 | | | | | X | | X | X | | | | X | X | X | X | X |
| | 4 | | | | | X | | X | X | | | | X | X | X | X | X |
| | 5 | X | | X | X | X | X | X | | | | | X | | | | |
| | 6 | X | | X | X | X | X | X | | | | | X | | | | |
| | 7 | X | | X | X | X | X | X | | | | | | | | | |
| | 8 | X | | X | X | X | X | X | | | | | | | | | |
| | 9 | X | | X | | X | X | X | | | | | | | | | |
| | 10 | X | | X | | X | X | X | | | | | | | | | |
| Earth Park- ing Orbit | 10a | X | | X | | X | X | X | | | | | | | | | |
| | 10b | | | | | X | X | X | | | | | | | | | |
| | 10c | X | | X | | X | X | X | | | | | | | | | |
| | 11 | | | | | X | X | X | X | X | X | X | | | | | |
| | 12 | | | | | | | | | | | | | | | | |
| | 13 | | | | | X | X | X | X | X | X | X | | | | | |
| | 14 | | | | | | | | | | | | | | | | |
| | 15 | X | X | | | | | | | | | | | | | | |
| | 16 | X | X | | | | | | | | | | | | | | |
| | 17 | X | X | | | X | X | X | X | X | X | X | | | | | |
| | 18 | | | | | X | X | X | X | X | X | X | | | | | |
| | 19 | | | | | | | | | | | | | | | | |
| | 20 | X | X | | | | | | | | | | | | | | |
| | 21 | X | X | | | | | | | | | | | | | | |
| | 22 | | | | | X | X | X | X | X | X | X | | | | | |
| | 23 | | | | | | | | | | | | | | | | |
| | 24 | X | X | | | | | | | | | | | | | | |
| | 25 | X | | | X | X | X | X | | | | | | | | | |
| | 26 | X | | | X | X | X | X | | | | | | | | | |
| | 27 | | | | | | | | | | | | | | | | |
| | 28 | X | | | X | X | X | X | | | | | | | | | |
| | 29 | | | | | | | | | | | | | | | | |
| | 30 | X | X | X | X | X | X | X | | | | | X | | | | |
| | 31 | X | X | | | | | | | | | | | | | | |
| | 32 | | | | | | | | | | | | | | | | |
| | 33 | | | | | X | X | X | | | | | X | X | X | X | X |

X Denotes significance.

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TABLE I-1-IV. APOLLO ENVIRONMENTAL PROFILE

ASSEMBLY & PRE-LAUNCH

| ENVIRONMENTAL FACTORS | | | | | | | | | | | |
|-----------------------|---|---|---|--|--|--|-----------------------------------|--|--|---|---|
| PHASE | DEFINITION | SHOCK | VIBRATION | PRESSURE | TEMPERATURE | GRAVITY | LOCAL WINDS | SALT SPRAY | SAND AND DUST | PRECIPITATION | HUMIDITY |
| 1 | Factory sub-assembly and transport | Worst conditions: free fall and pivoting drops during handling. Height based on size and weight per MIL-E-4970. Max. of 10 in. carry or tow negligible | Max. for carried equipment: $\pm 1.3g$ peak at 2-20 cps, 0.036 in. double amplitude at 20-52 cps, $\pm 5.0g$ peak at 52-500 cps | Ground transport: 15.4 to 10.2 psi (-1300 to 10,000 ft.). Air: 10.2 to 1.69 psi (10,000 to 50,000 Ft.). Rate of change: 1 in. Hg. Sec. | 24 hour cycle of max. temp. and solar radiation: 10 hrs. at 90 degrees F (no sunshine), followed by 5 hrs. temp. increase to 125 degrees F with rad. intensity increase to 105 watts 1 square foot, then 4 hrs. const. conditions and finally 5 hrs. temp decrease to 90 degrees F. Wind speed at 5 to 10 ft., 7 MPH | NOT APPLICABLE | | Exposure to salt spray environment of coastal region. Max. within 20 ft. of coast, 100 degrees F, 1001 mm diameter dust, 3 in MIL-E-55272C | Particle mean diameter between 0.15 and 0.30 mm. Concentration of 10 per foot width cross section up to 10 ft. with max. accumulation of 100 degrees F, 1001 mm diameter dust, 3 in MIL-E-55272C | Rain: 4 in. per hr. max. Hail: 1.2 in. dia. at 4 in. per hr. with 30 MPH wind | Exposure to 100% humidity, condensation in form of water or frost |
| 2 | Field assembly, checkout and installation | Bench handling (unpacked equipment) with 1g vertical for 1 in. flat drops, 4 in. rotational drops lateral (5-2000 cps) and 0.15g horizontal for 0.5 sec. 3 cps for 0.5 sec. duration. Booster mating: 1.2 in. free drop | Intermittent. Worst case during firing tests: Max. 4g long. and 2g lateral (5-2000 cps). | | Mainly sheltered in air conditioned environment. Intermittent exposure to outside ambient and solar radiation. Range: -10 degrees F to -125 degrees F | | | Mainly sheltered. Intermittent exposure to salt spray of coastal region | Mainly sheltered. Intermittent exposure to 4 in hr max. rain | Mainly sheltered. Intermittent exposure to 100% humidity | Mainly sheltered. Intermittent exposure to 100% humidity |
| 3 | Pre-launch countdown | | | 15.4 to 12.8 PSI | | Gravity gradient of -3.0g x 10-6 ft. sec ² ft. at level of 32.2 ft. sec | Up to 40 MPH with gusts to 60 MPH | | | Max. rainfall: 4 in. per hour for 30 min. | Same as Phase 1 |
| 4 | Launch count-down | | | | Same as Phase 1 | | | | Same as Phase 1 | | Same as Phase 1 |

LAUNCH & PRIMARY BOOST

| PHASE | DEFINITION | ACCELERATION | VIBRATION | ACOUSTIC NOISE | PRESSURE | DENSITY | TEMPERATURE | LOCAL WINDS |
|-------|---|--|---|--|--|--|--|-------------------------------------|
| 5 | Thrust buildup | Lift-off at 1.2g's | | Booster sound pressure level max. of 150db (referred to 1000 Hz) over range of 18-9600 CPS | Local Pressure 12.8 to 15.4 psi | 7.65 x 10 ⁻² lb/ft. ³ | Same as Phase 1 | Up to 40 MPH with gusts to 60 MPH |
| 6 | First stage boost | Up to 2.6g longitudinal. Max 1g lateral | | Booster sound pressure level decrease and boundary layer SPL from 140 to 130 db | From local pressure down to approx. .6 psi | Down to: 4.1 x 10 ⁻³ lb/ft. ³ | Ambient down to 390 degrees R. Aerodynamic heating negligible. | Max. of 60 MPH with gusts to 80 MPH |
| 7 | First stage thrust tail-off separate; ignition and thrust buildup | Max. of 2.5g's in min. of 1 sec. | Estimated maximum of: 4g's longitudinal 2g's lateral (5-2000 CPS) | SPL at 130 to 140 db | At: 0.6 psi | At: 4.1 x 10 ⁻³ lb/ft. ³ | Ambient at 390 degrees R. | |
| 8 | Second stage boost | Up to 5.2g longitudinal. Max. 1g lateral | | SPL dropping to negligible amount | Down to: 1.7 x 10 ⁻⁷ psi | Down to: 3.8 x 10 ⁻¹⁰ lb/ft. ³ | Radiant heating (See Phase 11). Aerodynamic heating: Transfer rate less than 5 BTU/ft ² - sec | |
| 9 | Second stage thrust tail-off separate; third stage chill-down, ignition, and thrust buildup | Max. of 5g's in min. of 1 sec. | | | At: 1.7 x 10 ⁻⁷ psi | At: 3.8 x 10 ⁻¹⁰ lb/ft. ³ | | |
| 10 | Third stage boost | Up to 2.5g long. Max. 1g lateral | | | Down to: .8 psi | Down to: 1.1 x 10 ⁻¹¹ lb/ft. ³ | | |
| 10a | Third stage boost | Up to 1.2g long. Max. 1g lateral | | | Down to: .8 psi | Down to: 1.1 x 10 ⁻¹¹ lb/ft. ³ | | |
| 10b | Earth Orbit | | NEGLIGIBLE | | At approx: 2.5 x 10 ⁻⁸ psi | At: 2.5 x 10 ⁻¹¹ lb/ft. ³ | Radiant heating (See Phase 11). | |
| 10c | Third stage boost to escape | Up to 2.5g long. Max. 1g lateral | Same as phases 5 through 10, 10a | | Down to: .8 psi | Down to: 1.1 x 10 ⁻¹¹ lb/ft. ³ | | |

TABLE I-1-IV. APOLLO ENVIRONMENTAL PROFILE (Continued)

CISLUNAR FLIGHT AND LUNAR ORBIT

| PHASE | DEFINITION | ACCELERATION | SHOCK | PRESSURE | DENSITY | TEMPERATURE | GRAVITY | IONIZING RADIATION | METEORIODS | GEOMAGNETIC FIELD |
|-------|--|---|---|---|---|--|--|---|---|--|
| 11 | Mid-course flight | None aside from Phases 20 and 21 | | From 1.3×10^{-8} psi to below 10^{-18} psi at 20,000 nmi (see Figure I-1-10). See Phase 18 for lunar atmospheric pressure. | From 1.1×10^{-11} to 10^{-12} lb./ft ³ below 10-22 lb./ft ³ at 20,000 nmi (see Figure I-1-10). See Phase 18 for lunar atmospheric density. | Thermal irradiance: see test for conditions. Solar Rad: 445 btu/hr-ft ² . Earth thermal rad: Max of 70.8 btu/hr-ft ² . Earth Albedo rad: Max of 160 btu/hr-ft ² . See phase 18 for Moon thermal and albedo radiation. | Earth gravity gradient decreasing from a max. of 3×10^{-6} ft/sec ² ft to 0 ft/sec ² ft. See Phase 18 for lunar gradient. | Cosmic Radiation: Primarily protons, max dose rate approx. 25 milliroentgens/hr-ft ² . Ionizents above a given altitude: see Phase 18. Solar Flares: Max of 12 per yr. upper July 1959-1960. Max of 3 per yr. peak value proton flux est as: N(E)- 10^{10} -5 proton/cm ² -sec-max; Van Allen rad: see Table I-1-V. | See Figure I-1-15. 'Nucleon' effects per square meter per second with meteoroids above a given altitude: see Phase 18. 'Mass in grams'. | Max. of 0.6 gauss. Max. of 0.6 gauss. of altitude and latitude. See Figure I-1-16. |
| 12 | Extend space communications devices | NONE | NEGLECTIBLE | | | | | | | |
| 13 | Lunar reconnaissance | | NOT APPLICABLE | | | | | | | |
| 14 | Mission module release if required | NONE | NEGLECTIBLE | | | | | | | |
| 15 | Course corrections, as required | Up to 5 corrections, 1/2g to 2g, 0.5 to 1.5 sec. each. | | | | | | | | |
| 16 | Initial injection into high altitude lunar orbit | 2g or less over full phase (about 75 secs. or more) | Rise time minimum of 25 msec to full acceleration | | | | | | | |
| 17 | Computation and orbital correction to refined lunar orbit | At end of phase 25 sees or more of 2g or less | | | | | | | | |
| 18 | Low perilune orbit, scientific observations and data acquisition | None Aside From Phases 20 and 21 | | | | | | | | |
| 19 | Mission module release, if required | NONE | NEGLECTIBLE | | | | | | | |
| 20 | Injection maneuver into earth return trajectory | 2g or less over full phase duration (about 100 secs. or more) | Rise time minimum of 25 msec to full acceleration | | | | | | | |
| 21 | Course corrections, as required | Up to 5 corrections, 1/2g to 2g, 0.5 to 1.5 sec. each | | | | | | | | |
| 22 | Mid-course flight | None Aside From Phase 24 | | | | | | | | |
| 23 | Staging to re-entry configuration | NONE | NEGLECTIBLE | | | | | | | |
| 24 | Course corrections, as required | Up to 5 corrections, 1/2g to 2g, 0.5 to 1.5 sec. each | Rise time minimum of 25 msec to full acceleration | | | | | | | |

NOT SPECIFICALLY APPLICABLE - CONTAINED IN PHASE 11.

IN EARTH MOON ENVIRONMENT AS SPECIFIED IN PHASES 11 AND 18

NOT SPECIFICALLY APPLICABLE - CONTAINED IN PHASE 11.

NOT SPECIFICALLY APPLICABLE

NOT SPECIFICALLY APPLICABLE - CONTAINED IN PHASE 18

SAME AS PHASE 11

NOT SPECIFICALLY APPLICABLE - CONTAINED IN PHASE 22

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TABLE I-1-IV. APOLLO ENVIRONMENTAL PROFILE (Continued)

RE-ENTRY, DESCENT, AND RETARDATION

| PHASE | DEFINITION | ACCELERATION | ACOUSTIC NOISE | PRESSURE | DENSITY | TEMPERATURE |
|-------|-------------------------|---|--|---|---|---|
| 25 | Pullout | 12 g max., above 6g for max of 30-50 seconds | Boundary layer sound pressure level max of 140 db | Between 2.6×10^{-7} and 3.3×10^{-3} psi | Between 7.6×10^{-10} and 2×10^{-5} lb/ft ³ | Max heat transfer rate 500 to 2000 btu/ft ² -sec at stagnation point, diminishing to 500 btu/ft ² at 100 ft alt. Most windward meridian of afterbody, and down to 100 to 500 along most windward meridian of forebody |
| 26 | Cross range maneuvering | Max. of 3g and slight increase due to lifting devices | Boundary layer SPL max of 130 db | Between 2.2×10^{-3} and 4.9×10^{-3} psi | Between 1.4×10^{-5} and 2.8×10^{-5} lb/ft ³ | |
| 27 | Deploy lifting devices | | NOT SPECIFICALLY APPLICABLE - CONTAINED IN PHASES 26 AND 28 | | | |
| 28 | Equilibrium glide | Max. of 3g | Boundary layer SPL max of 120 db | To between 0.10 and 1.7 psi | To between 1.2×10^{-3} and 1.2×10^{-7} lb/ft ³ | Buildup of integrated flux. See Phase 30 |
| 29 | Terminal guidance | | NOT SPECIFICALLY APPLICABLE - COMPLETELY CONTAINED IN PHASES 28 AND 30 | | | |

LANDING

| PHASE | DEFINITION | ACCELERATION | SHOCK | VIBRATION | ACOUSTIC NOISE | PRESSURE | DENSITY | TEMPERATURE | LOCAL WINDS | SALT SPRAY | SAND AND DUST | PRECIPITATION | HUMIDITY |
|-------|--|---|--|--|--|---|--|--|---------------------------------------|------------|---------------|---------------|----------|
| 30 | Terminal maneuver and let-down | Max of 3g except for Phase 31 (acceleration and shock), 1g after Phase 31 | Negligible except for Phase 31 and touch-down impact max. of 20g for 0.2 secs. | Possible low frequency instabilities and buffeting in transonic region | Boundary layer sound pressure level max. of 130 db | From min. of 0.16 psi to local ground pressure (see Phase 33) | From min. of 10^{-3} lb/ft ³ to local ground density (see Phase 33) | Integrated flux buildup to 70,000 to 190,000 btu/ft ² at stagnation point, diminishing to 20,000 along most windward meridian of afterbody, and to 2,000 to 40,000 along most windward meridian of forebody | Up to 80 MPH with gusts up to 125 MPH | | | | |
| 31 | Deploy terminal retardation devices. | Transition from 3g max. (steady state) to 1g. See shock | Max peak of approx 9g's for max of 15 secs | NOT SPECIFICALLY APPLICABLE - CONTAINED IN PHASE 30 | | | | | | | | | |
| 32 | Extend impact attenuation devices | NEGLECTIBLE EFFECT | | | | | | | | | | | |
| 33 | Recover and secure craft. Disembark crew | NEGLECTIBLE | | | | | | | | | | | |
| 34 | Debriefing | ENVIRONMENT NOT APPLICABLE | | | | | | | | | | | |

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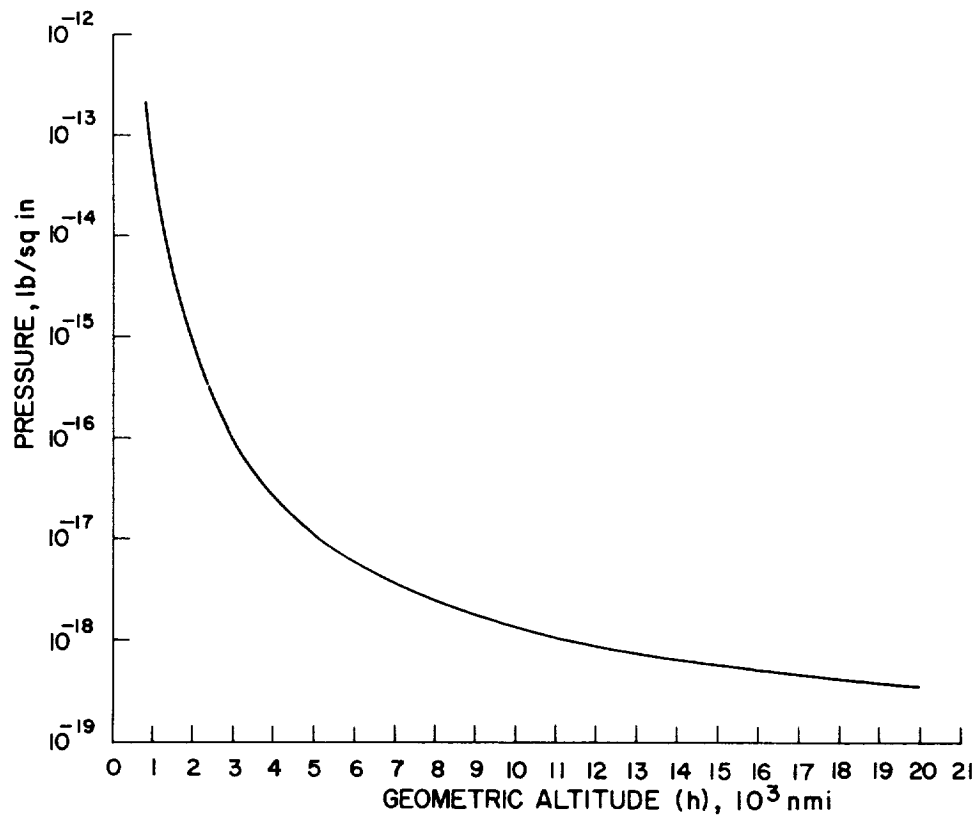


Figure I-1-10. Atmospheric pressure for earth - extended ARDC, 1959

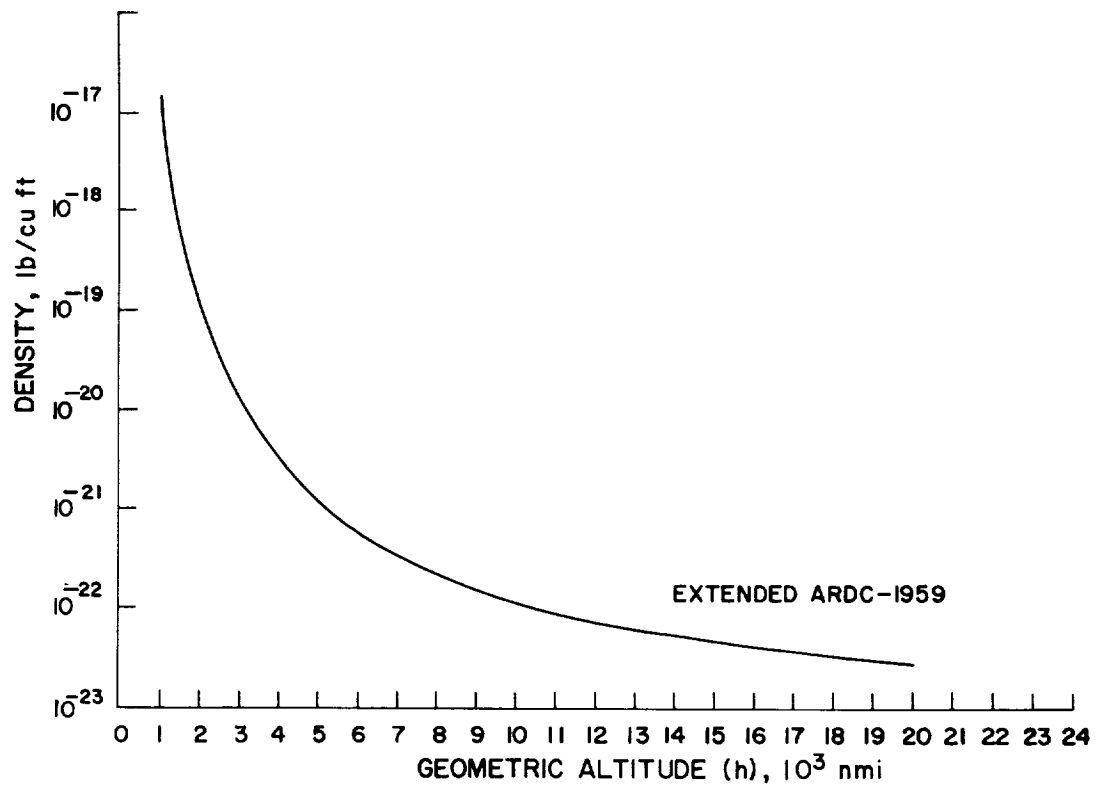


Figure I-1-11. Atmospheric density for earth - extended ARDC, 1959

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In the vicinity of the Moon, a minor lunar atmosphere (pressure and density) can be considered. One estimate of this is given in reference 5 as less than 10^{-12} earth's atmosphere at sea level.

1.3.4 Temperature

Temperature or heat flux is an important environmental factor through all mission phases. The estimates for ground conditions are based primarily on data contained in reference 6. During powered boost, the transition is made from ground ambient to cold, black, space where radiation flux contributes to vehicle temperature. Also, during powered boost, aerodynamic heating occurs. The estimate of this given in Table I-1-III is based on the shallow ascent trajectory of the Saturn booster. For re-entry, estimates are given for maximum heat transfer rate and integrated heat flux for a nominal range of re-entry conditions.

In space, thermal irradiance from the sun, earth, and Moon are typified as vectors having magnitude and direction. From reference 7, a good average value for solar radiation flux in the earth-Moon vicinity is 445 BTU/hr-sq ft. This will be directed from the sun to a vehicle surface normal to the rays.

With average earth temperature taken as 250 degrees K (450 degrees R), essentially constant, and assuming an emissivity of 1.0, the earth radiant heat loss is 70.8 BTU/hr-sq ft. Data on the Moon is much less certain. Estimates can be obtained from reference 5 which sets the temperature in the range from 374 degrees K at full sunlight to 120 degrees K at darkness. These would correspond to moon radiant heat loss of 26.4 BTU/hr-sq ft and $0.28 \frac{\text{BTU}}{\text{hr-sq ft}}$ respectively. The earth and Moon fluxes will be directed outward along a radius from the body to the vehicle. The values given must be modified to account for the size of the earth (or Moon) disk viewed from the vehicle (function of altitude) and the angle between the earth (or Moon) radius intercept with the vehicle and the normal to the incremental vehicle surface. These considerations are contained in a configuration factor, F_e . A plot of this is shown in Figure I-1-12 which is derived from reference 8. The factor, F_e , is applicable to both earth and Moon.

An average value for earth albedo can be taken as 0.36 (reference 7); and for the Moon, 0.073 (reference 5). These are equivalent to 160 and 32.5 BTU/hr-sq ft respectively. From reference 8, the factor F_a which modifies these maximum values is approximated by:

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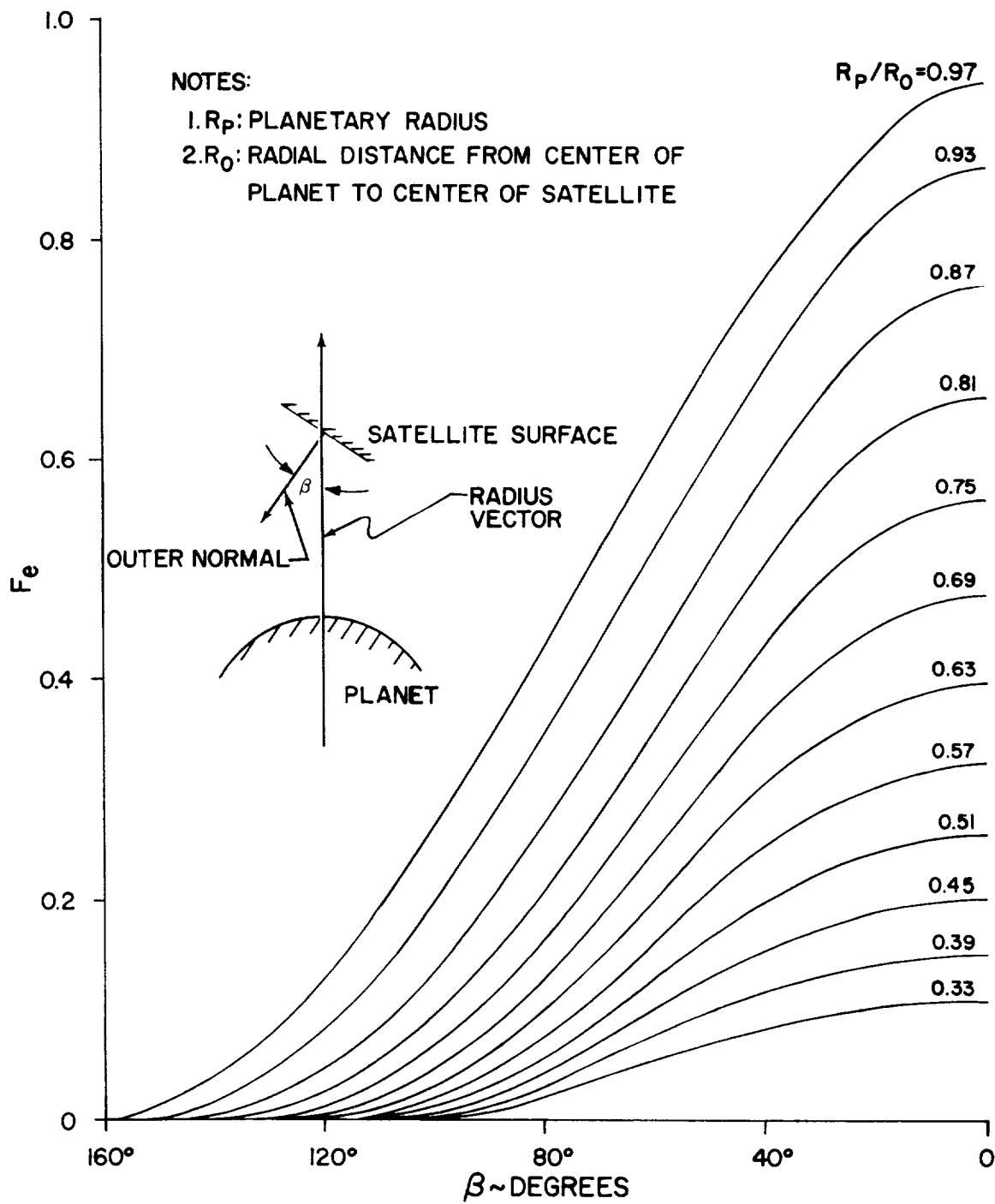


Figure I-1-12. Configuration factor F_e between satellite surface and spheric planet

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$$F_a = F_a \left(\frac{R_o}{R_p} \right) \cdot \cos \alpha \cdot F(\beta)$$

where $F(\beta) = \frac{F_e}{F_e \text{ (at } \beta = 0)}$ (From Figure I-1-12),

α is the angle between the sun-planet center line and the vehicle-planet center line. $F_a \left(\frac{R_o}{R_p} \right)$ is plotted in Figure I-1-13. The factor F_a can refer to Moon or earth.

1.3.5 Gravity

The absolute value of gravity is not applicable to the mission either on the ground or in flight. The gravity gradient can be of importance where ultrasensitive inertial instrumentation is used in free spaceflight and mutually referred elements are located a distance apart along on earth or Moon radial. The gradient for the earth or Moon can be obtained from the following expression (derivative of the gravitational field function):

$$\text{Gradient (Moon or earth)} = -2g_o \frac{R_B^2}{(R_B + h_B)^3}$$

where g_o = acceleration due to gravity (g_o earth = 32.2 ft/sec²)

at surface of body (g_o Moon = 5.3 ft/sec²)

R_B = radius of body R_B earth = 2.09×10^7 ft.

R_B Moon = 0.571×10^7 ft.

h_B = altitude above surface of body

1.3.6 Ionizing Radiation

Cosmic radiation is generally isotropic and uniform in space except as influenced by magnetic fields or atmospheric absorption. (Composition: Protons, > 90% in number, alpha particles about 7%). Energy content has been recorded up to 10^{17} ev, with the average about 3.6×10^9 ev. Figure I-1-14, from reference 2 gives the estimated cosmic radiation dose rate at four latitudes.

Solar radiation bursts have a maximum occurrence of about 12 per year and a conservative estimate for the violent solar proton bursts of May and July 1959 would be a maximum of 3 per year. The peak value of proton flux may be approximated by:

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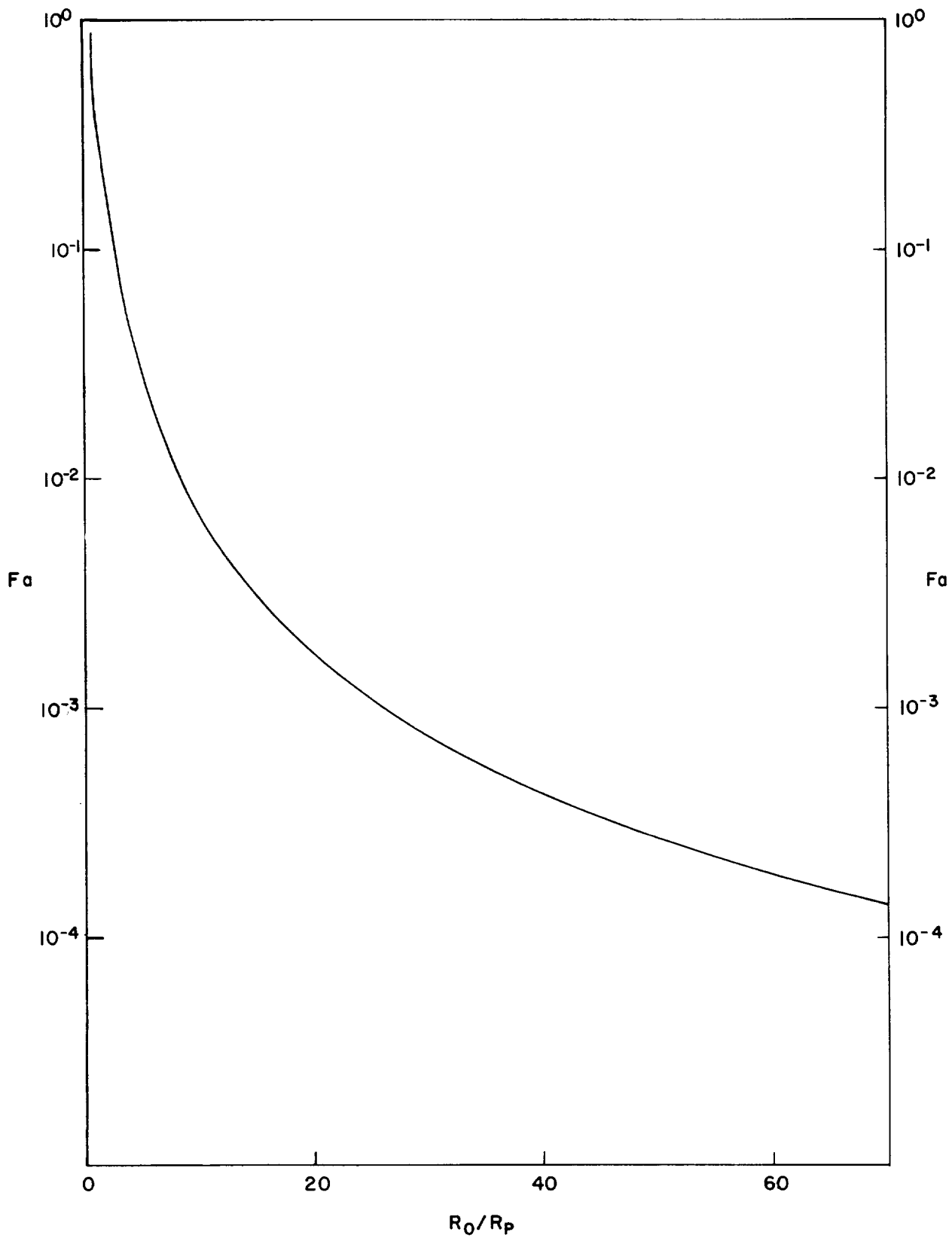


Figure I-1-13. F_a (albedo factor) vs. radius of orbit / radius of planet

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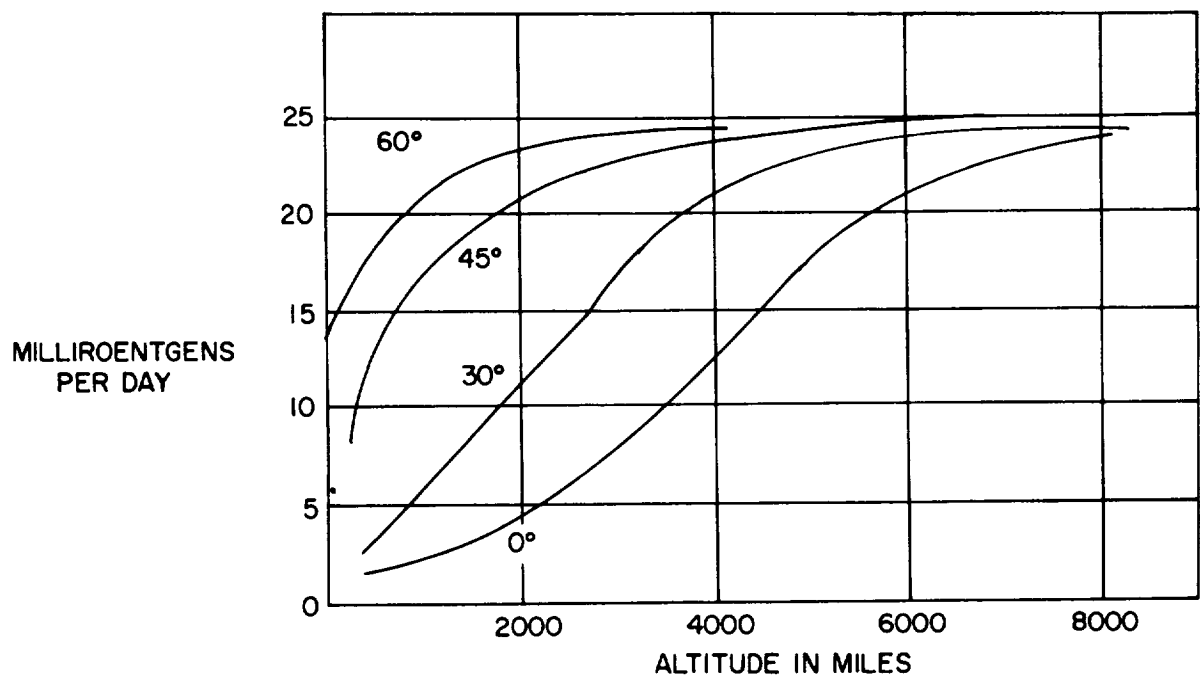


Figure I-1-14. Estimated cosmic radiation dose rate at four latitudes

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$N(E) = 10^{10} E^{-5}$ protons/cm²-sec-MEV for E above 10 MEV. $N(E) = 0$ for E below 10 MEV). The average flux over a several month period will be less than one percent of the peak value. (Reference 9)

A summary of estimates of radiation in the peaks of the "inner" and "outer" Van Allen belts is contained in reference 9. Table I-1-V presents data from this reference. The interaction of space radiation and the APOLLO design is covered in Volume V, Human Factors.

1.3.7 Meteoroids

Estimates of the meteoroid population have been made by a number of workers in the field and are summarized in reference 4. Figure I-1-15 taken from reference 4, gives the number of impacts per square meter of exposed area per second expected to occur with sporadic meteoroids of a given mass or greater, versus mass in grams. Also shown in figure 6 for convenience are approximate values of magnitude and depth of penetration in aluminum. A certain amount of directivity could perhaps be noted since meteoroids travel in orbit about the sun.

1.3.8 Geomagnetic Field

The geomagnetic field is significant during near-earth space flight, and not especially significant during ground, boost, or re-entry phases. The primary effects of this field would be on instrumentation. The intensity of the dipole field as a function of altitude and latitude can be approximated by the expression.

$$H = .311 \left[\frac{R_e}{R_e + h} \right]^3 \left[\frac{1}{1} + 3 \sin^2 \theta \right]^{1/2} \text{ gauss}$$

where θ is the latitude measured from the magnetic equator. For convenience, this is plotted in Figure I-1-16.

1.3.9 Local Winds

Local winds are of significance during: (a) pre-launch and launch countdown, when alignment of booster and instruments is accomplished; (b) the lower altitude portion of boost where intensity of the wind may be large and heavy perturbations to the boost vehicle dynamics or trajectory could result; (c) the lower altitude portion of the return, where terminal dynamics could be seriously affected; and (d) after touchdown, where recovery could be inhibited. The values shown in Table I-1-IV are derived from reference 6.

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TABLE I-1-V.
SUMMARY OF IONIZING RADIATION IN SPACE ENERGY LEVEL VS. FLUX DENSITY

| ALTITUDE (FROM EARTH'S SURFACE), NAUTICAL MILES | ELECTRON FLUX | | PROTON FLUX | | DATA SOURCE | NOTE |
|--|---------------|---|--------------|--|-----------------------|---|
| | ENERGY LEVEL | NO. OF ELECTRONS Per cm ² -sec. | ENERGY LEVEL | NO. OF PROTONS Per cm ² -sec | | |
| 600 n. miles (Approx.) | 200 Kev | ~2 x 10 ⁶ (1) | 100 Mev | 4 x 10 ² (2) | (1) M. Walt. et al. | AT 46° N. Lat. |
| | 500 Kev | ~5 x 10 ⁴ (1) | 400 Mev | 10 ¹ (2) | (2) Freden & White | |
| 2000 n. miles (VAN ALLEN "INNER ZONE" — PEAK — Equatorial Orbit) | > 20 Kev | ~2 x 10 ⁹ (3) | >40 Mev | ~2 x 10 ⁴ (3) | (3) Van Allen | (Values est. within factor of 5) Data Extrapolated |
| | 600 Kev | ~10 ⁷ (3) | <40 Mev | <2 x 10 ⁴ (4) | (4) Freden & White | |
| Alt. range estimated; 400-3500 miles. | >20 Kev | 2 x 10 ¹⁰ (PEAK) | >10 Mev | <10 ⁷ | A. Rosen — S. T. L. | |
| | | 1 x 10 ¹⁰ (AVG.) | | | | |
| >200 Kev | | 1 x 10 ¹⁰ (PEAK) | >40 Mev | 2 x 10 ⁴ | | |
| | | 5 x 10 ⁷ (AVG.) | | | | |
| >600 Kev | | 1 x 10 ⁷ (PEAK) | | | | |
| | | 5 x 10 ⁶ (AVG.) | | | | |
| > 20 Kev | | ~10 ¹⁰ - 10 ¹¹ (5) | >75 Mev | <2.0 (8) | (5) Van Allen, Vernov | (a)no data available - based on max. trappable flux. (b)No. of protons (10-75 Mev) assumed equal to electrons (E>500 Kev) (c)Measured values. Other values calculated |
| >200 Kev | | ~10 ⁸ | | | (6) Rosen | |
| >500 Kev | | ~10 ⁶ (7) | <75 Mev | NO DEFINITE DATA | (7) Vernov | |
| >5000 Kev | | <1.0 (7) | | | (8) Simpson, et al. | |
| 19, 400 n. miles (EQUATORIAL; AVERAGE DATE (9)). | 0 - 50 Kev | 3 x 10 ¹⁰ (a) | 0-2 Mev | 10 ¹⁰ (a) | (9) R. Rochlin | |
| | 200 Kev | 5 x 10 ⁸ | 2-10 Mev | 2 x 10 ⁹ (b) | | |
| | 500 Kev | 3 x 10 ⁷ (c) | 10-25 Mev | 10 ⁷ (b) | | |
| | 1000 Kev | 8 x 10 ⁶ | 25-75 Mev | 10 ³ (c) | | |
| | 5000 Kev | 3 x 10 ⁵ | >75 Mev | 0 (c) | | |
| | 13, 000 Kev | 4 x 10 ⁴ | | | | |
| >13, 000 Kev | 0 | | | | | |

Numbers in parenthesis are reference designations taken from:

Riethof, T. R., "Charged Particle Radiation in Space", General Electric TIS Report R60SD391, August 1960.

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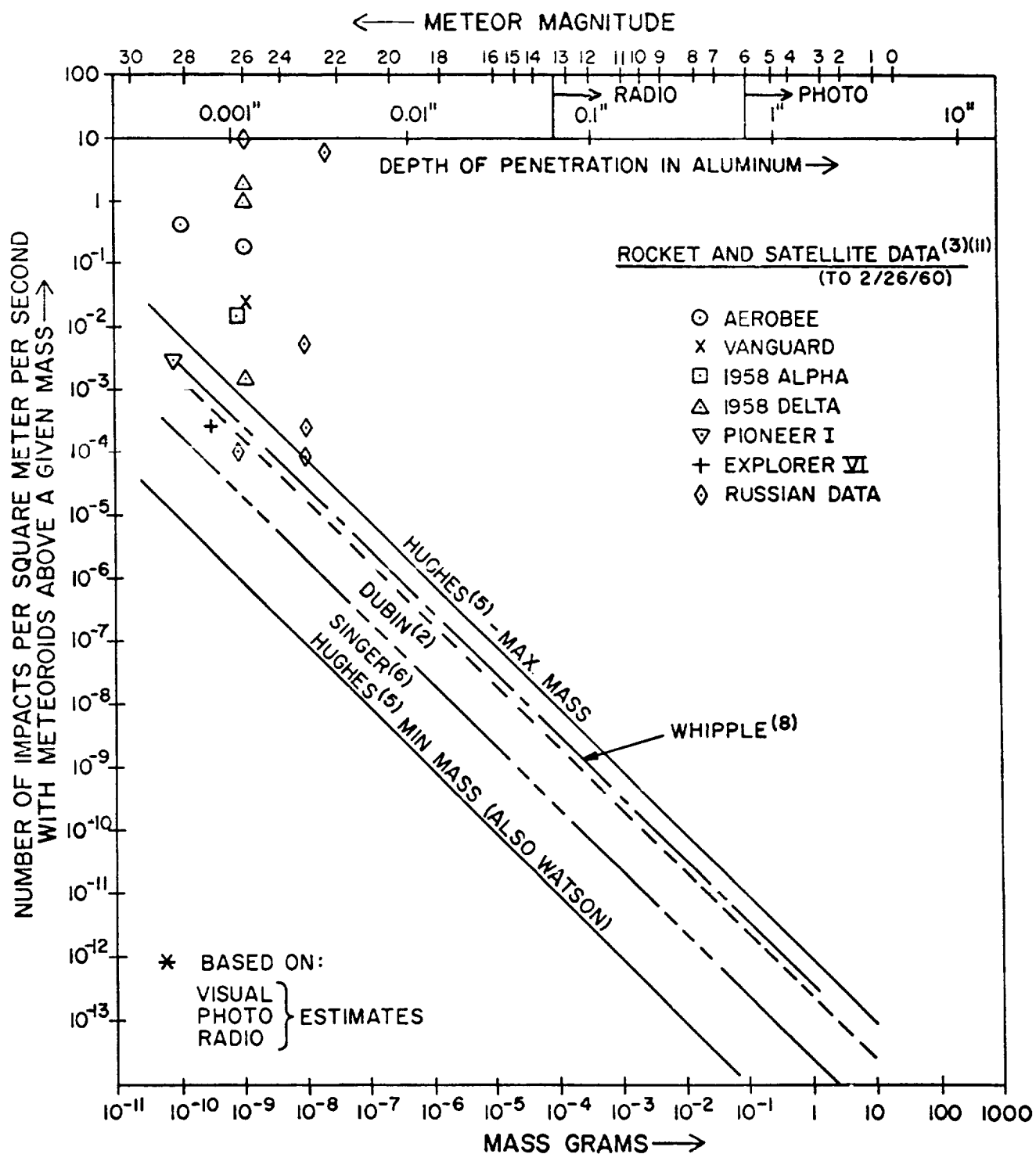


Figure I-1-15. Population of sporadic meteoroids*

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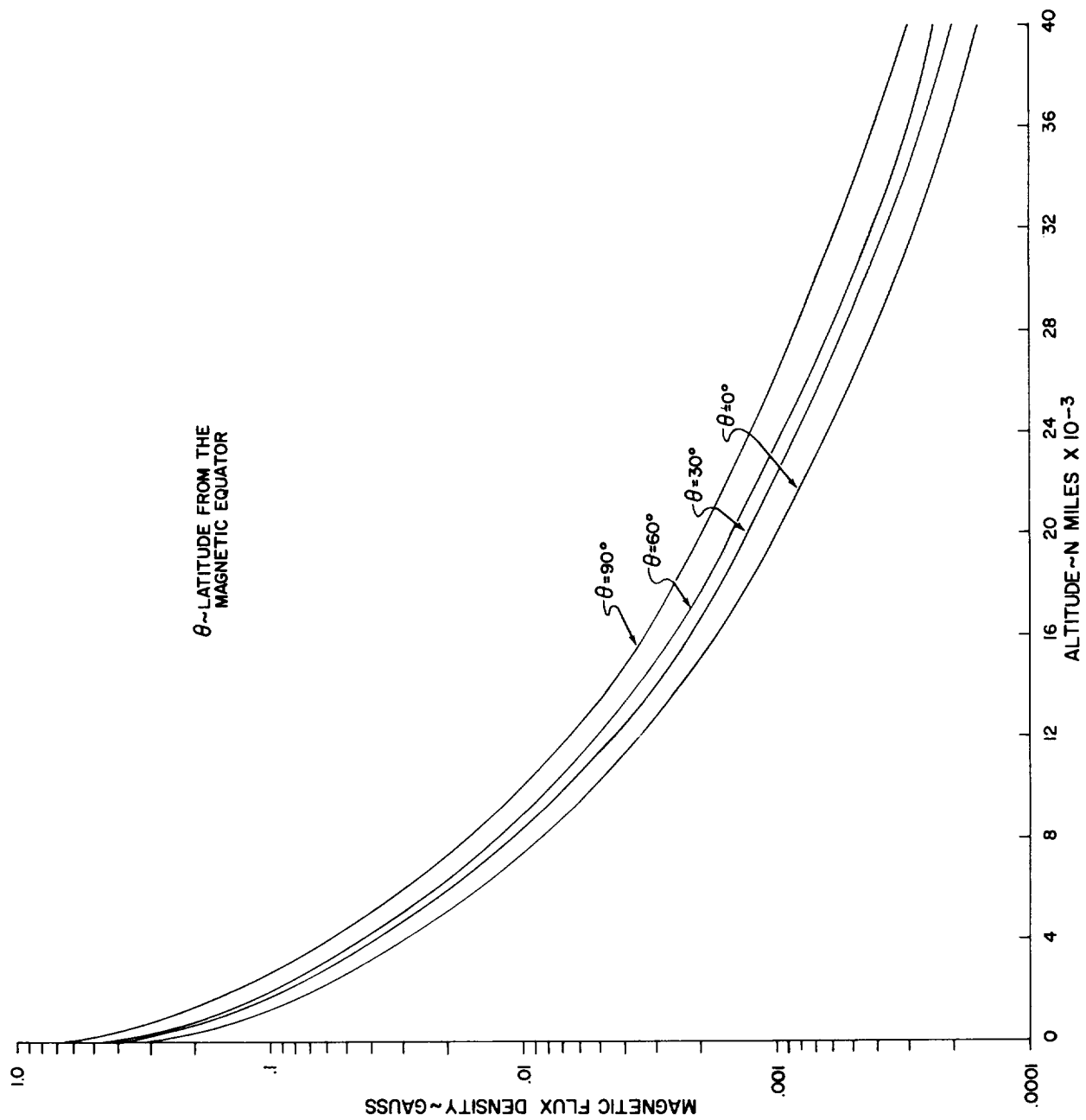


Figure I-1-16. Altitude vs. geomagnetic field intensity

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1.3.10 Salt Spray, Sand and Dust, Precipitation, and Humidity

These factors are present in the atmosphere at altitudes above ground; however, the exposure times during powered boost and re-entry are comparatively short. Their significance will be limited to location on the ground (pre-launch and post-touchdown). The maximum environment for sand and dust, humidity, and precipitation are based on MIL-STD-210A, reference 6. Landing may be on land or water.

1.3.11 Environment References

1. MIL-E-4970A (USAF), Military Specification: Environmental Testing, Ground Support Equipment, General Specification for.
2. U. S. Air Force Specification Bulletin No. 523, Space Environmental Criteria for Environmental Vehicles, 28 November 1960.
3. Franken, P. A., and Kerwin, E. M., Jr. "Methods of Flight Vehicle Noise Prediction", WADC TR 58-343 (AD205776), November, 1958.
4. Handbook of Satellites and Space Travel, General Electric 58SD131, 18 April 1958.
5. Bobrovnikoff, N. T., "Natural Environment of the Moon", Ohio State University Research Foundation, WADC Phase Technical Note 847-3, June, 1959.
6. MIL-STD-210A, Military Standard: Climatic Extremes for Military Equipment, 2 August 1957.
7. Costello, F., and Latour, A. P., "Subsystem L (SARV) Mark I Heat Balance During Flight, General Electric, MSVD Aerophysics Engineering Technical Memorandum #127, 30 January 1959 (SECRET).
8. Beretsky, I., "Orbital Heat Flux Calculations", General Electric, MSVD. Aerophysics Engineering Information Release ATE-117-029, 6 January 1960.
9. Riethof, T. R., "Charged Particle Radiation in Space", General Electric TIS Report R60SD391, August 1960.

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1.4 EMERGENCY PROCEDURES

The emergency procedures are defined here as all those which do not follow the planned mission. These procedures may simply involve in-flight repair or use of subsystem redundancies, and have no serious effect on the mission. They may be of a reprogramming nature, which could modify (shorten) the mission, where the emergency is not one of impending disaster or where the vehicle was beyond the "point of no prior return"*.

Finally, the procedures may involve where required, true aborts, for booster escape, or quick-return trajectories. Modified mission and abort programs are keyed to nominal abort programs are indicated, and will be discussed below; however, the detailing of these procedures has been the object of specific studies which are reported in Chapter III of this volume.

Abort Programs I through IV are all characterized by an abort boost to permit escape from the primary booster. The abort boost required to provide escape while on the pad or soon after first stage ignition (Abort Program I) will not loft the escape vehicle high enough to permit any significant maneuvering during descent. Abort Program II will result in atmospheric flight at altitudes and velocities adequate for maneuvering during descent, but not requiring re-entry precautions. Abort Program III will result in ballistic flight to re-entry (IIIa) or an emergency earth orbit (IIIb) and subsequent re-entry. Program IIIa will be used where immediate return is required. Typically, Program IIIb will be used where escape from the primary booster is the only emergency requirement and an earth-parking orbit can be used for a modified mission and return to a favorable landing site. Program IV will result in super-orbital velocities where a return maneuver will be required for proper direct re-entry (IVa) or an intermediate earth orbit (IVb). Program V, occurring after release from the third-stage booster, will not require an abort boost as such, but rather, a return maneuver to a direct re-entry (Va) or to an intermediate orbit (Vb).

On the Normal/Emergency Mode Relationship Chart, Figure I-1-17, modified missions are indicated by dashed lines; they will generally involve reduced capability of the vehicle and/or subsystems. Manual emergency operation and repair will be a probable requirement, and the mission module will probably be released prior to the scheduled time.

*The "point of no prior return", if it exists, is defined here as that point of time, occurring during cislunar flight towards the Moon, beyond which an earliest safe return would be obtained by proceeding with a circum-lunar pass and direct re-entry.

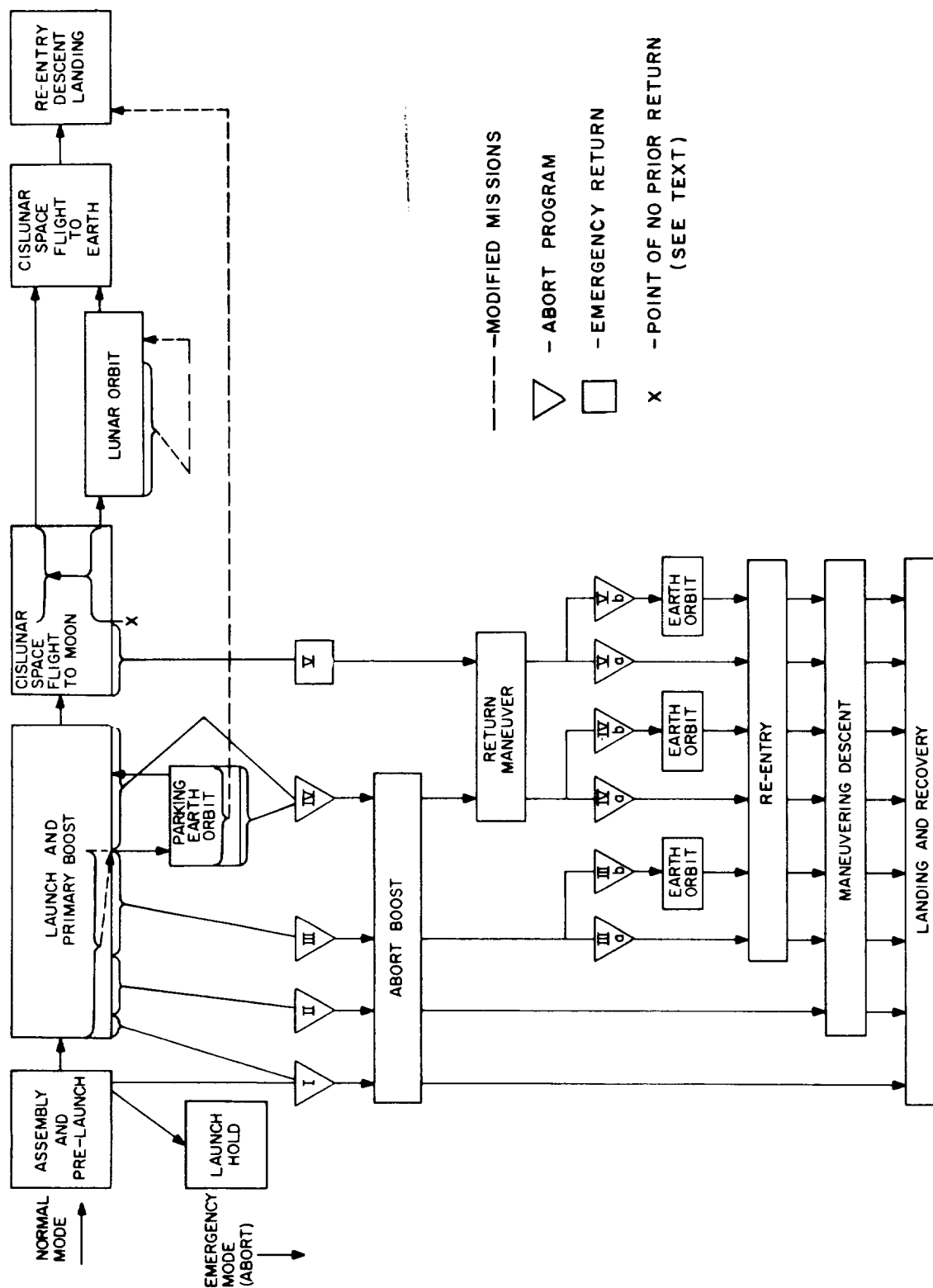


Figure I-1-17. Normal/emergency mode relationship

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During the primary boost phase, a non-abort emergency requirement is expressed as a programming change to provide injection into an Earth orbit for subsequent return, thereby short-circuiting the entire lunar mission. Mission modification during the cislunar-to-Moon phase after passing the point of no prior return (X), would consist of reprogramming the lunar orbit mission to a circumlunar pass. Mission modification during the lunar orbit is expressed as a shortening of the in-orbit phase. After injection into the return trajectory, no earlier-than-normal returns are possible.

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2.0 Preliminary Landing Site Selection Studies

The landing site selection studies have an objective similar to the mission profile analysis: to provide preliminary design guidelines. The intent here is to pinpoint some of the considerations which are pertinent to the terminal aspect of the mission. Except for some highly untenable circumstances (e.g., political and climatic), site selection from the ground facilities standpoint cannot be made independently of trajectory considerations. Within each of these two aspects, there are feasibility/economic tradeoff elements which must be further compared between the two before an optimized trajectory/site criterion is established. The effort reported here is a preliminary approach to the problem as an initial step in a necessarily iterative process.

Through considerations of politics, climate, accessibility, tracking and likely trajectories, some preliminary selections have been made including priority assignment and criteria for operational selection.

2.1 COARSE SELECTION

A number of criteria may be established and used to select gross areas of interest in which favorable landing sites may be located. The task of establishing criteria for refined selection will be made simpler if it is applied only to those areas which pass a preliminary screening based on certain coarse requirements. Care must be taken, however, to avoid the indiscriminate application of any coarse criteria such as to exclude particular small areas or sites which may have unique characteristics which would otherwise recommend them.

Considerations of climate, politics, and accessibility are useful in the reasonable reduction of the area of interest. Climatic considerations are important from the standpoint of temperature, precipitation and fog. The last two factors cannot be generalized at this point; they are regionalized too finely, and will be considered later. Temperature is important if a three-day survivability after landing is to be guaranteed. Also, extreme temperatures would inhibit the recovery operations. A reasonable limit would be to constrain the landing site between 50 degree North and 50 degree South latitudes. This latitude

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band will generally include the tropical and mesothermal climates, ensuring at least eight months of over 34 F. An additional constraint will be imposed in the North Atlantic with a latitude bound of 40 degrees North due to a rough prevailing sea state. These bounds would appear to exclude no sites of any particular advantage.

Political considerations would lead us to exclude the land areas of Soviet influence, as the first step. In addition, in order to avoid political complications, all of the remainder of the Asia mainland will be excluded. Though of a different nature, political complications would also suggest the exclusion of certain friendly countries. Prominent among these, due to large population density and/or potential inflammatory reactions are Europe, Japan, Central America, Mexico (excluding Baja California), and the West Indies (excluding Puerto Rico, the Lesser Antilles and the Bahamas).

From the standpoint of inaccessibility, the East Indies, the Philippines, South America, and Africa will be excluded. The generally rugged features of these areas could seriously inhibit search and recovery operations in the event of a non-nominal landing. The Sahara Desert, although not of rugged terrain, is excluded due to its general inaccessibility and torrid climate, which could seriously limit chances for survival and inhibit recovery operations. However, on this basis, the Great Australian Desert is not excluded due to the Woomera missile-range facilities. The exclusions made to this point are indicated by the darker shaded areas on the map in Figure I-2-1.

Considerations of search and recovery forces staging base possibilities lead to further exclusions. These are shown as three lighter shaded areas in Figure I-2-1. These areas roughly enclose locations beyond the reasonable travel distance from feasible staging bases. This distance varies from perhaps 700 to 1500 nautical miles depending on staging location. The remaining areas in Figure I-2-1 generally consist of three zones (Atlantic Missile Range, North-South Pacific range along South America, and the broad Pacific area between Australia and the United States) and continental United States. These areas may be considered as the sum total of sites for which reasonable search and recovery potential is available. The following discussions will be concerned more with the reduction to the nominal landing sites location.

2.2 TRAJECTORY CONSIDERATIONS

Further refinement of the selection can be made through consideration of the nominal mission trajectory. There will be a dependency of one upon the other such that complete independent refinement of either would not appear to be possible. Specification of the

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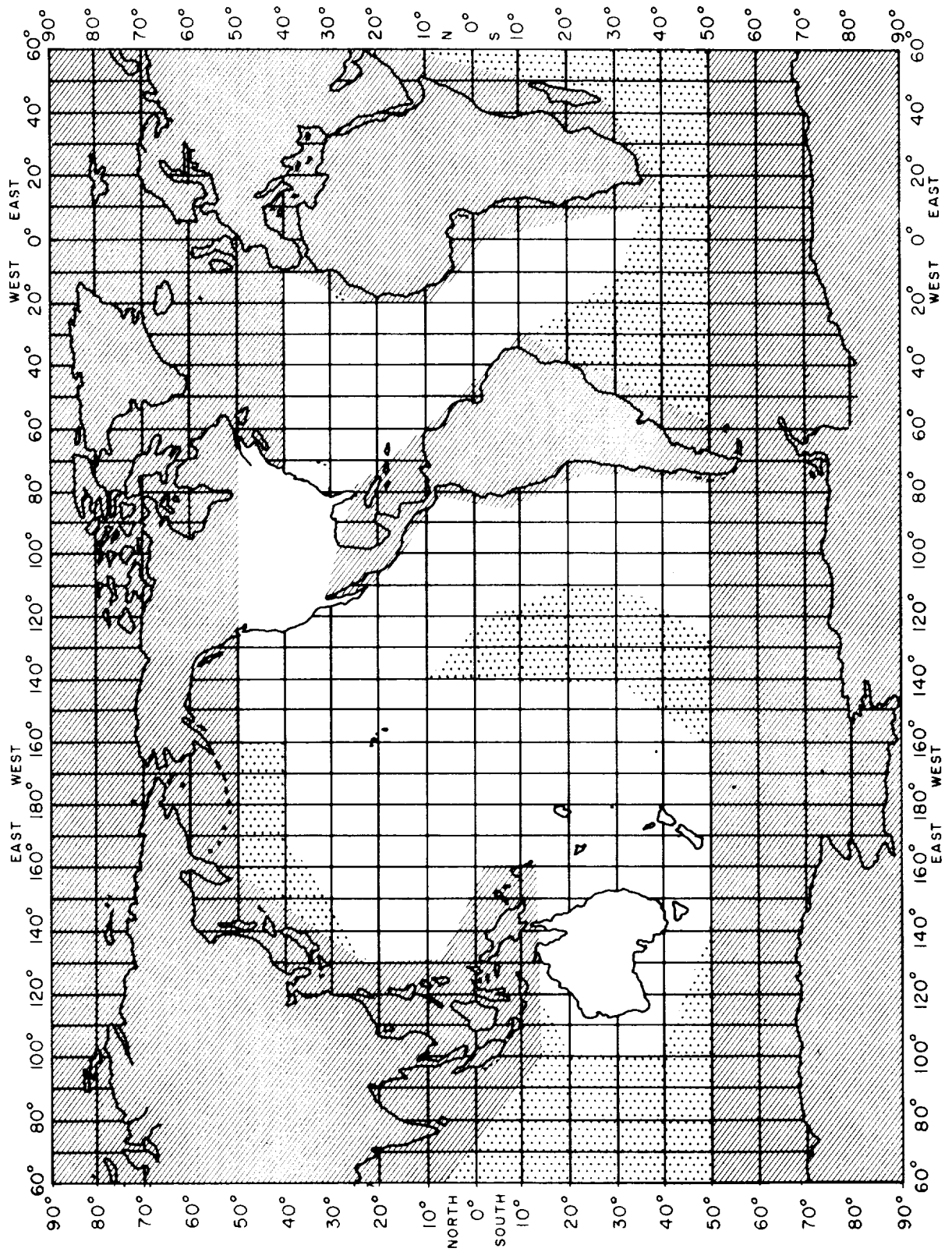


Figure I-2-1. Coarse selection of landing areas

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outbound trajectory would not clearly establish the return due to the significant change in inclination which could occur during the swing around the Moon. Therefore, for our purposes here, we will merely consider the return orbit to be direct (generally West to East). This consideration will be of value in the determination of tracking constraints on site selection.

A further trajectory consideration which can be made is the sensitivity of a non-maneuvering return to errors in position (latitude and longitude) at time of re-entry or to orbital inclination. Specification of return orbit and re-entry parameters to give a minimum cross range error for a non-maneuvering return could supply a trade-off parameter for selection of the nominal landing site and the nominal return trajectory.

2.3 TRACKING CONSIDERATIONS

Tracking of the re-entry vehicle is critical if search and recovery time is to be minimized. On the return mid-course trajectory, once inside the high-percentage coverage altitude of the deep space network, a Mercury-type net must take over the tracking assignment. This net would most desirably consist, first of all, of present Mercury facilities. In addition, limited ship-borne tracking could be added at critical points, existing tracking or communications sites given increased tracking capability, or finally, additional sites installed. Obviously, the last should be avoided if possible. A final tracking capability which should be available is the lower altitude terminal facility. Whether the landing site is on land or water, permanent or temporary, terminal tracking should be provided to aid in the approach maneuver to a nominal landing.

Terminal tracking considerations from the standpoint of non-interference (as opposed to facilities), will favor landings on or near water and on land with a flat land approach. From the retained areas of Figure I-2-1, we can consider some of these possibilities. Water landing areas in all three zones are possible of course. The islands of the Atlantic would qualify for near water landings (from a tracking, not landing terrain standpoint) as well as those of the Pacific along the western shore of South America and in the broad Pacific zone (including Hawaii, New Zealand, and Tasmania). The western shore of the United States and Baja California are excellent possibilities. Due to the approach paths which would be required, the northern and southern shores of Australia would appear unfeasible; the western Australian shore remains a possibility. Although they would appear to be of adequate size, the Great Lakes regions are eliminated for either on or near water siting due to the heavy population and cultural buildup in the vicinity.

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For land siting, the great bulk of the United States does not appear feasible since an adequate terminal tracking range could result in a trajectory extending eastward from the Rockies well into the central lowlands with increasing population and cultural buildup. The southwestern United States region, including such facilities as White Sands Proving Grounds and the Las Vegas Gunnery Range is discounted because of the approach tracking limitations imposed by the Sierra Nevada, Rocky, and Sierra Madre Mountains. On the other hand, the flat land approach over Australia to the Woomera range impact area remains a possibility. On a preferential basis, terrain limitations would discount the previously mentioned islands for land siting.

Based on the preceding discussion, we can consider some representative landing cases. The nominal return trajectory through the atmosphere will be projected on a non-rotating earth's surface as an arc of a great circle. Earth's rotation during the return is distinguished by the gradual "capture" of the vehicle by the earth's atmosphere. The geographical track on a rotating earth for a West-to-East return will be shortened in longitude as a function of flight time. Range and time of flight through the atmosphere will be a function of the L/D ratio, and are discussed in Volume III. An illustrative geographic track, including earth's rotation is indicated on Figure I-2-2 for typical return of the APOLLO with the selected D-2 re-entry vehicle, to a typical site. Also indicated on Figure I-2-2 are existing Mercury and deep-space tracking facilities. Prime consideration is given here to the Atlantic Missile Range, Edwards Air Force Base, and the Woomera Range from the standpoint of facilities and near-terminal tracking. For the Atlantic Missile Range, the approach, as shown, is along the Range, for Edwards, the approach is from the sea, and for Woomera the approach is over a large flatland area. These factors will aid in the near terminal tracking. The Atlantic Missile Range has the advantage of Range tracking and established sites and recovery operations. Edwards has the advantage of tracking facilities at Point Arguello, Hawaii, Woomera/Muchea, and PMR ships. The Woomera approach would be dependent upon the Grand Canary facilities, the Zanzibar facilities and Indian Ocean ships and likewise would be restricted to low L/D if the long blackout period over mid-Africa is to be avoided. Hawaii could be a feasible alternate for water landings, with tracking from Johannesburg, Indian Ocean ships and a possible tracking installation on Guam.

The Atlantic Missile Range, especially since it will have the capability for launch abort recovery, would be quite feasible for the nominal site.

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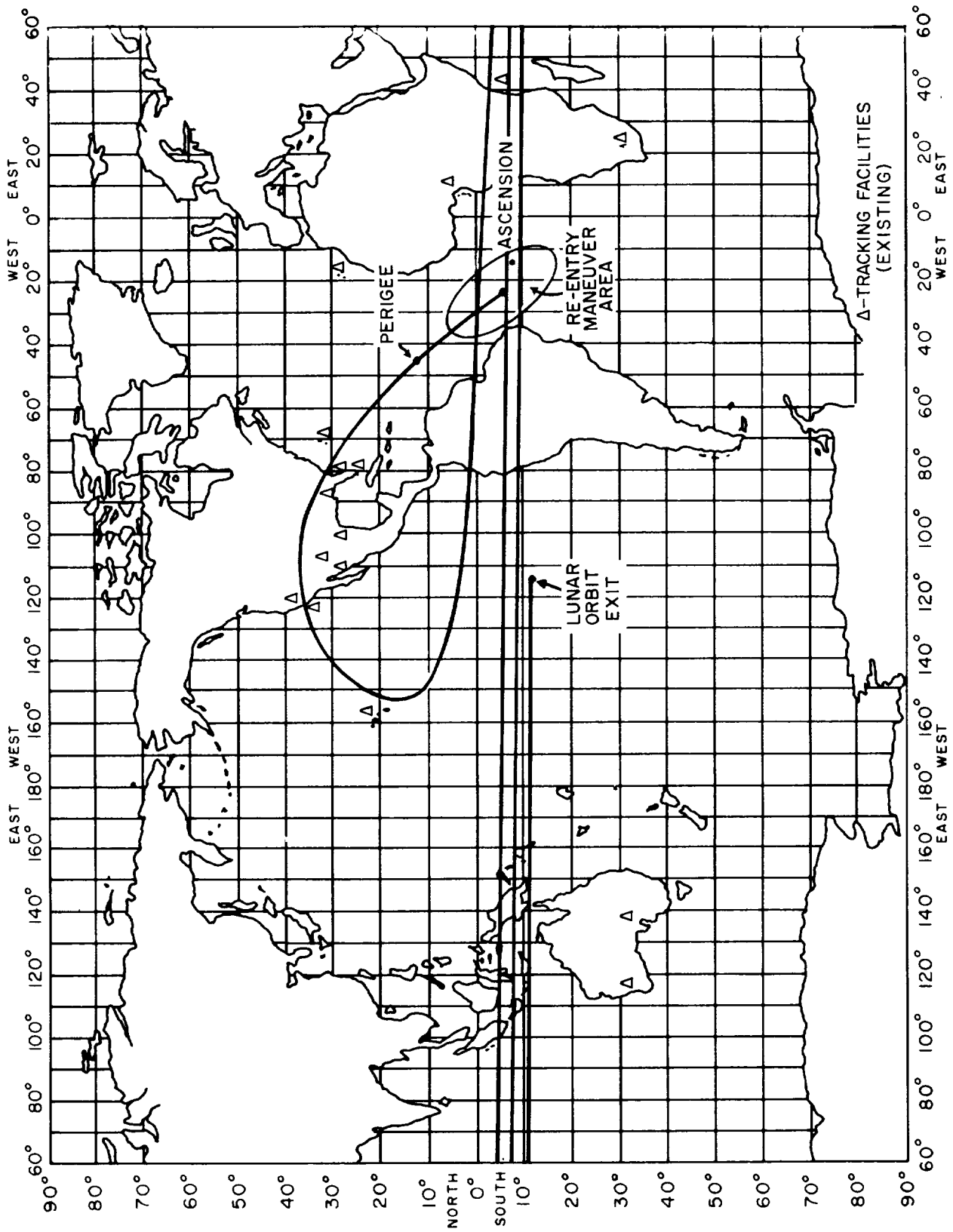


Figure I-2-2. Illustrative return trajectory no. 1

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CHAPTER II

RELIABILITY

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CHAPTER II

RELIABILITY

2.1 INTRODUCTION

Since design for performance alone will inevitably sacrifice reliability, feasibility of the APOLLO design is a function of reliability as well as performance. As the APOLLO system moves from the feasibility stage through research and development to manufacture and, finally to operational use, the evolution of the system must be accompanied by an integrated reliability program which furnishes data that are fed into the design in the same manner as performance data. Just as inadequate performance requires design change, so does insufficient reliability. The type of reliability program required for Project APOLLO is outlined in Volume IX, "APOLLO Program Implementation Plan." In the pursuit of the design evolution performance criteria have not been, and will not be, allowed to overrule reliability criteria (where trade-offs are applicable), due to default in the factoring-in of reliability considerations. Other trade-offs with reliability, such as development cost and time, must be acknowledged and evaluated as they occur. That is, care must be taken at every step in the design to avoid the blind sacrifice of reliability.

A positive approach to reliability has been taken in the preliminary design presented in the volumes of this report. Throughout the course of the study, subsystem and system preliminary design has been based on continual analysis of environmental factors and requirements imposed by the elements of the system complex on each other. Preliminary estimates of reliability have been used where applicable to pinpoint problem areas and to provide a basis for decision among competing design approaches. Potential failure areas have been analyzed and the design adjusted to minimize their effects. The preliminary system design presented in the volumes of this report has been based on a deep concern for reliability; it provides a design approach having a potential for high probability of both crew survival and mission success.

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2.2 THE SYSTEM COMPLEX

By definition, reliability is the probability of success. In the APOLLO context, success must be measured not only in terms of completion of the objective mission but also in terms of man (crew) survival. It is necessary to consider man as part of the system complex. Although this presents problems in assuring the survival of man, as well as that of the equipment, the presence of man in the complex can be advantageous in enhancing the operational reliability of the overall system. Man can contribute to the system reliability in several ways. He can exercise decision-making capability as necessary throughout the mission. He can also recognize equipment functional degradation and take action to correct performance drift before deviations occur beyond acceptable limits. Furthermore, in the event of certain equipment failures, maintenance and repair action can be carried out. These possibilities can result in attainment of reliability that would not be possible in an unmanned system designed to perform a similar task.

Man must be protected from surrounding stresses and an acceptable artificial environment maintained. This necessitates special environmental protection -- passively, by means of structure and devices, and actively, through an environmental control subsystem. In the main, this environmental protection will also reduce the strain on the operational equipment. The relationships between crew and the external environment, and between equipment and the external environment, are shown in Figure II-2-1. Pictorially, these two relationships differ slightly; actually, in terms of required and designed-in protection, they differ markedly.

The reliability of the overall system complex is dependent upon man, the environmental control equipment, and the operational equipment. Man can provide for active control and maintenance of both the environmental and operational equipment. The environmental control equipment provides environmental protection for man and stress protection for the operational equipment. The operational equipment supports the environmental control equipment while providing for the accomplishment of the overall mission. This is depicted in Figure II-2-2. These three vital elements make up an integral equipment loop in assuring mutual survival and adequate functioning of the system complex. Equipment reliability is a necessity in providing an artificial environment for man. Man can contribute toward this relationship by enhancing equipment reliability through maintenance action.

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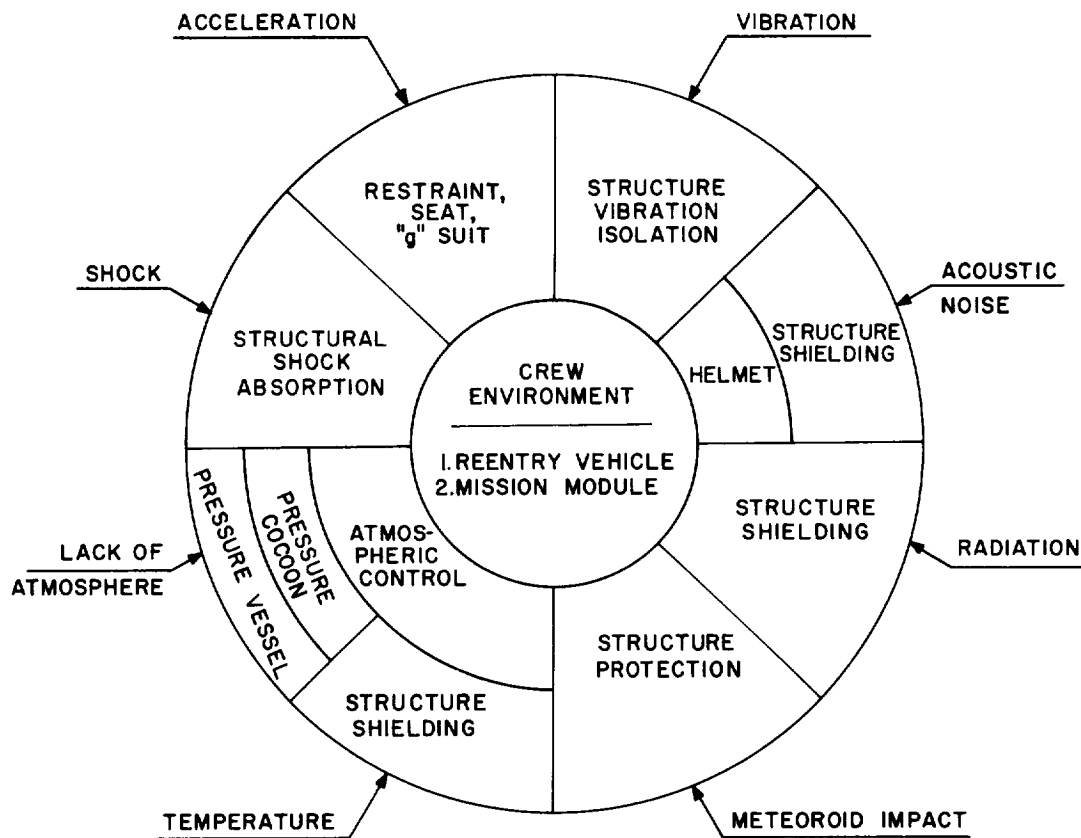


Figure II-2-1a. Relationship between crew and the external environment

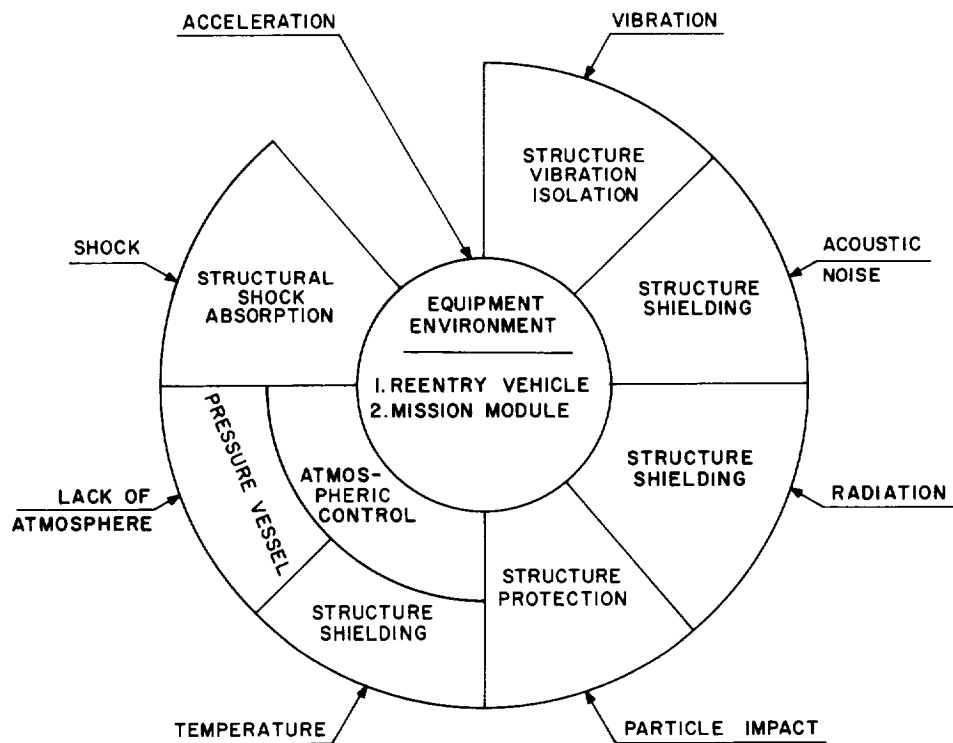


Figure II-2-1b. Relationship between equipment and the external environment

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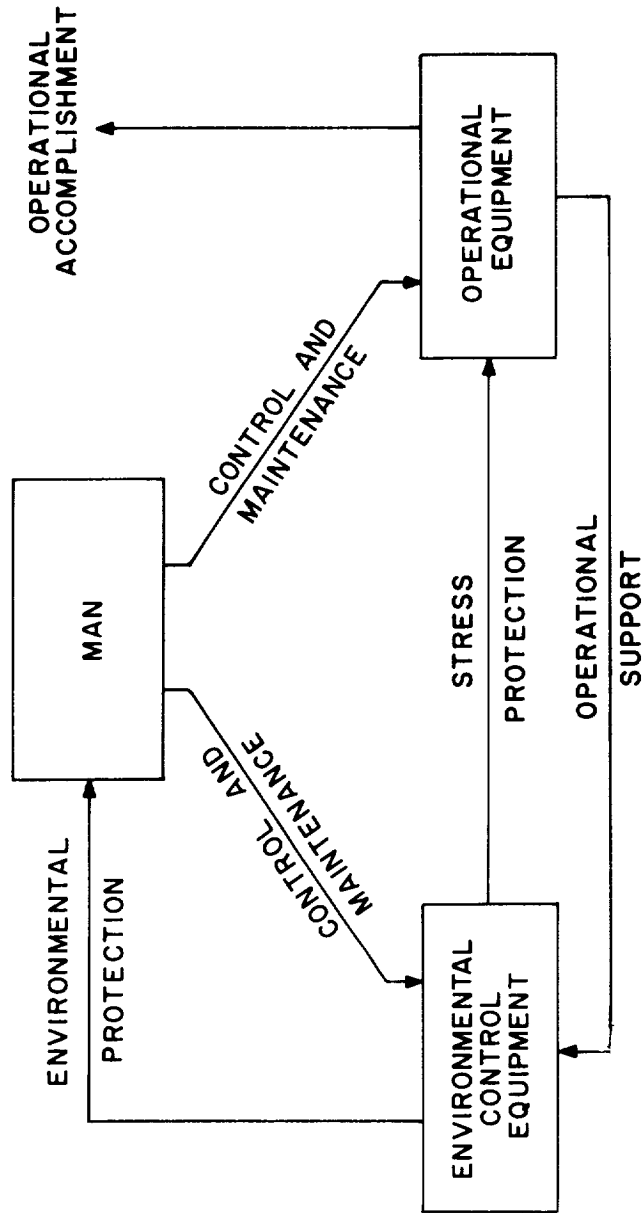


Figure II-2-2. System complex

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2.3 DEFINITION OF SUCCESS

For the manned APOLLO mission, success must be defined both in terms of the mission accomplishment and in crew survival. Of course, these two aspects are not strictly independent; there can be no mission accomplishment if there is no crew survival. On the other hand, the converse is not necessarily true; crew survival is but one aspect of, and does not ensure, complete mission accomplishment. Further, as we shall see, higher probability of crew survival can be achieved than probability of mission accomplishment. For these reasons, crew survival and mission accomplishment, although not independent, are treated as individual aspects.

Mission accomplishment must be measured in terms of meeting mission objectives whether the mission is developmental (e.g. an unmanned off-the-pad abort) or an operational lunar reconnaissance orbit. The mission will accomplish less if a departure from the scheduled mission is required. Departure from the scheduled mission can take many forms, ranging from the relatively minor variations (e.g. that due to failure of the scientific instrumentation) to major variations such as emergency returns. These variations may be classified as alternative missions, and they will have lesser objectives to meet. Meeting the lesser objectives still provides a measure of success, which, although less than that for the primary missions, salvages the results from absolute failure.

Providing the capability for mission alternatives can be important from the standpoint of investment in time and money in attaining some mission objective(s).

When viewed from the standpoint of crew survival, mission alternatives are a critical requirement. Consider Figure II-3-1, illustrating, schematically, the effect of emergency mode capability on the probability of crew survival. By means of emergency-mode backup, total probability of crew survival can be brought to an acceptable level.

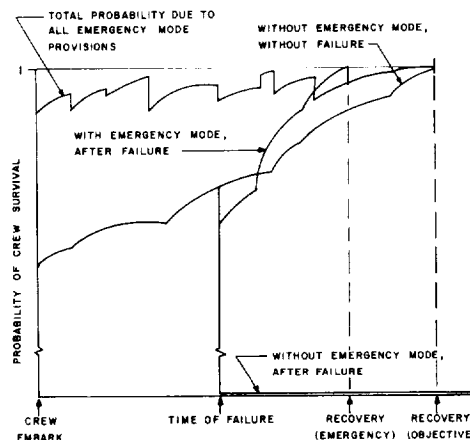


Figure II-3-1. Effect of mission emergency mode capability on probability of crew survival

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2.4 RELIABILITY GOALS

Reliability goals are an important factor in the APOLLO Program objectives. The advanced requirements for system performance embodied in the APOLLO mission do not permit the assumption of adequate reliability -- it must be a design factor. The question at the moment is, "What reliability goals should be set for the twin aspects of mission accomplishment and crew survival?"

The preliminary design, as detailed for the selected configuration D-2, contains sufficient redundancy, alternative and backup modes of operation, and provides for such a high order of crew participation (in the form of maintenance, repair and monitoring) that the reliability goal for crew survival in the ultimate operational system has been set at 99 percent. By the same token, mission accomplishment, aside from Saturn probability of successful launch, has a reliability goal of 90 percent.

During the preliminary design study, the approach toward implementing the above goals has been the adoption of techniques which offer the greatest potential for ensuring crew survival and mission success.

2.5 RELIABILITY ASSURANCE IN DESIGN

As discussed above, reliability has been provided initially for in the equipment design. Several design approaches, as shown in Table II-5-I, have been followed, at least broadly, if not in depth, to maximize system reliability attainment. As the design matures, the accompanying reliability program pursues these approaches more deeply.

2.5.1 Circuit Element Optimization

One of the first design approaches that must be taken is the reduction of system complexity. This can be accomplished through design simplification within the parameters mutually established by reliability, performance, accuracy, and other system requirements.

The utilization of redundancy is a valuable method for improving the operational reliability of a system. In the selected configurations, the duplication of equipment has been employed to increase the performance reliability of critical elements. For instance, the multiple rocket abort system has been designed to provide crew escape from the booster in the event of imminent explosion. Such a maneuver requires only seven solid rockets of the size used. The system, however, has been designed with eight rockets to ensure survival. As with

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any redundancy, this design resulted in a weight debit which was traded for a reliability credit. The additional weight in this case is 228 pounds.

The selection of the most reliable parts available for the application is obviously of utmost importance in designing to achieve high reliability. This has been accomplished by acquiring a knowledge of the past reliability history of the considered part; and, where possible, selecting a component part that has proven its capability in a similar application. Such an approach is demonstrated in the D-2 landing system, in which the parachutes utilized have an enviable record of accomplishment. Where part data history is unavailable, the considered parts will be subjected to a thorough engineering evaluation to assure operational suitability. A high-reliability parts selection program of particular application to APOLLO is in effect on Advent. This program, described in Volume IX, includes 100 percent screening and control during manufacture and 100 percent component tests under complete space environmental conditions as well as extended acceptance tests in vacuum. Under such a program, improvements in part failure rate can be projected for APOLLO application. A listing of these is presented in Table II-5-II.

The stresses to which component parts are subjected are a function of application, environment, and operating conditions. The strength of the part will vary as a function of time. To prevent catastrophic failure of component parts, the stress/strength distribution should never be allowed to overlap. The technique of part derating can be utilized effectively to reduce the operational failure rates of the circuit elements. Design operating conditions can be selected so as to reduce the power, or other load, on a part to a chosen percentage of the manufacturer's rating. The design percentage of the rated load can be determined from part application data for the particular part, considering its operational usage. A typical example of derating as applied to APOLLO lies in the power amplifiers used in both the telemetry and voice communication system. For application to the APOLLO design these amplifiers have been derated to about 30 percent of rated power. Application guides, presenting data for generally utilized component parts, have been issued by many companies and are available for analytical use. Part application data can also be compiled through a specifically designed test program. Packaging can be employed to minimize environmental stresses on circuit elements. The reduction of environmental stress on parts can be accomplished through the use of external cooling, vibration isolators, and similar packaging techniques. Adequate packaging consideration can effectively reduce the operational failure rate of a multiplicity of circuit elements, and in conjunction with the other design techniques provide optimum system reliability.

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TABLE II-5-I TECHNIQUES IN DESIGNING FOR MAXIMUM RELIABILITY

Circuit Topology Optimization

- (1) Reduction of Equipment Complexity
- (2) Utilization of Redundancy

Circuit Element Optimization

- (1) Selection of Best Parts for the Application
- (2) Utilization of Derating Factors
- (3) Packaging to Minimize Environmental Stresses

Biotechnological Optimization

- (1) Human Use Factors
- (2) Maintenance Provisions
- (3) Psychobiological Elements

TABLE II-5-II. PROJECTED* FAILURE RATES IN PERCENT/1000 HOURS
FOR
APOLLO ELECTRONIC COMPONENTS

PART TYPE

| | |
|-------------------------|--------|
| Batteries (per cell) | 0.05 |
| Bearings | 0.002 |
| Bolometers | 0.075 |
| Capacitors | |
| Ceramic | 0.001 |
| Glass & Vitreous enamel | 0.0005 |
| Mica | 0.0005 |
| Paper | 0.0005 |
| Tantalum | 0.005 |
| Variable Air | 0.009 |
| Variable Ceramic | 0.008 |

* Conservative (i.e. slightly higher than Minuteman data)

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TABLE II-5-II. PROJECTED* FAILURE RATES IN PERCENT/1000 HOURS
FOR
APOLLO ELECTRONIC COMPONENTS (Continued)

| <u>PART TYPE</u> | <u>FAILURE RATES</u> |
|------------------------------|----------------------|
| Choppers | 0.031 |
| Circuit Breakers | 0.005 |
| Clutches | 0.006 |
| Connectors | |
| Multi-pin (free flight) | 0.0001 per pin |
| (Ground & Powered Flight) | 0.003 per pin |
| r.f. (free flight) | 0.002 per pin |
| (Ground & Powered Flight) | 0.006 per pin |
| Crystals | 0.004 |
| Electron Tubes (per section) | |
| Diodes | 0.026 |
| Klystrons | 0.120 |
| Magnetrons | 7.500 |
| Microwave Switching | 0.280 |
| Rectifiers | 0.270 |
| Thyratrons | 0.025 |
| Triodes, Pentodes | 0.053 |
| Voltage Regulators | 0.012 |
| Filters (mechanical) | 0.014 |
| Fuzes | 0.010 |
| Gyroscopes | 0.085 |
| Gears | 0.001 |
| Heaters | 0.001 |
| Inductors | |
| Power & Audio | 0.001 |
| r.f. & i.f. | 0.0005 |
| Saturable Reactors | 0.012 |
| Jacks | 0.0002 |
| Lamps | 0.100 |
| Meters | 0.050 |
| Microwave Components | |
| Delay Lines | 0.008 |
| Ferrite Cores | 0.050 |
| Loads & Attenuators | 0.015 |
| Power Ferrite Devices | 2.500 |
| Tuned Stubs & Cavities | 0.010 |

* Conservative (i.e. slightly higher than Minuteman data)

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TABLE II-5-II. PROJECTED* FAILURE RATES IN PERCENT/1000 HOURS
FOR
APOLLO ELECTRONIC COMPONENTS (Continued)

| <u>PART TYPE</u> | <u>FAILURE RATES</u> |
|--------------------------------|----------------------|
| Photocells | 0.075 |
| Relays | |
| General Purpose | 0.100%/10,000 cycles |
| Latching | 0.100%/10,000 cycles |
| Power | 0.150%/10,000 cycles |
| Sensitive | 0.220%/10,000 cycles |
| Thermal | 0.120%/10,000 cycles |
| Resistors | |
| Composition | 0.001 |
| Film | 0.0005 |
| Wirewound | 0.003 |
| Variable Composition | 0.004 |
| Variable Wirewound | 0.065 |
| Rotating Devices | |
| Motors | 0.015 |
| Dynamotors | 0.115 |
| Generators | 0.040 |
| Synchros & Resolvers | 0.002 |
| Semi-Conductor Diodes | |
| Germanium | 0.023 |
| Silicon | 0.002 |
| Solenoids | 0.004 |
| Stepping Switches | 0.128 |
| Switches | |
| Rotary | 0.118%/10,000 cycles |
| Sensitive | 0.045%/10,000 cycles |
| Toggle | 0.015%/10,000 cycles |
| Terminals, Joints, Connections | |
| (Free Flight) | 0.00005 |
| (Ground & Powered Flight) | 0.001 |
| Thermistors | 0.020 |
| Thermostats | 0.002 |
| Timers | 0.079 |
| Transformers | |
| Audio | 0.001 |
| Filament | 0.013 |
| Magnetic Amplifier | 0.002 |
| Power | 0.013 |
| Pulse | 0.007 |
| r.f. & i.f. | 0.004 |
| Transistors | |
| Germanium | 0.010 |
| Silicon | 0.004 |
| Vibrators | 0.040 |

* Conservative (i.e. slightly higher than Minuteman data)

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2.5.2 Biotechnological Optimization

The design approaches necessary to maximize the operational reliability of the space system complex must consider human usage, ease of maintenance, and psychobiological factors. While human use factors and maintenance provisions are self-evident, it should be noted that man will be confined in a small area in a strange environment for long durations without tangible work, and this will have an effect on his ability to function. Thus, the assignment of what man believes to be decision-making tasks and possible maintenance action could alleviate the psychological strains caused by extended periods of idleness in his new surroundings. The human element must be fully integrated with equipment engineering in order to attain an adequate system design.

2.5.3 Design Programs For Reliability Assurance

1. A key concept which would be applied to the APOLLO design during the development program is that of achieving a high level of system operational availability through sympathetic design. Sympathetic design can be described briefly as the practice of standardizing units of similar function in different subsystems so that they can be interconnected to provide mutual redundancy. This "mutual redundancy" effect can be accomplished by three basic techniques: (1) a modified form of straight redundant design, (2) the multifunctional block design method, and (3) the deliberate overdesign (aside from derating) of members of component families having similar and/or related functions.

The sympathetic design approach seriously considers the feasibility and practicality of designing an amplifier in one subsystem "in sympathy", it might be said, with an amplifier in one or more other subsystems. For instance, what are the trade-offs involved in expanding a little on band pass characteristics of a particular RF amplifier (or amplifiers) in the communication subsystem so that it could be substituted--either automatically or manually--for the RF amplifier in a radar-altimeter subsystem, such that the radar-altimeter subsystem could operate for some period of time (continuously or intermittently) at the same performance or at a permissible reduced performance? Further possible gains by such sympathetic design would obtain a considerable degree of subsystem redundancy and effective spares that would otherwise be prohibitive because of weight limitations and the following of usual subsystem design practices. A detailed

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example of the sympathetic design approach by the General Electric Company is presented in Reference 1. Analyses such as these during the APOLLO design phase may prove fruitful in achieving optimal reliability.

2. A further technique which would be used in designing emergency modes and maintenance capability into the APOLLO equipment is modal analysis. This technique is discussed in Appendix SC-C, and will be applied to the APOLLO design in association with redundant design, sympathetic design, and/or lesser mode operation.

2.6 RELIABILITY CONSIDERATIONS DURING PRELIMINARY DESIGN

An idealistic approach to reliability during a preliminary design study could start with an overall set of objective reliability figures and proceed with a reliability synthesis. This synthesis would include an apportionment of probabilities among the mission phases and the included phase-dependent subsystems. To attempt this apportionment of goals in a realistic manner among the normal (series) mission phases and also among initially assumed emergency modes would be a monstrous undertaking. This approach would require a substantial amount of iteration between the apportionment and mission/subsystem preliminary design.

The approach taken here is more practical, and does not inhibit progress in the preliminary design. On the basis of company and subcontractor experience in aircraft, missile, and satellite design, first approximations were made to the preliminary design of the system in accordance with the objective mission. Also included were estimated requirements for emergency modes based primarily on considerations for crew survival. (Factored-in here were the known data on the Mercury Program). Design criteria included pessimistic data relative to the environmental factors, trajectory calculations, guidance accuracies, fuel energy management, and extent of potential booster failures. Further iterations in preliminary design were directed towards reduced complexity and utilization of redundancy to increase reliability, and consideration of subsystem failure modes and abnormal operations, including emergency mode decisions. Crew contributions to performance and reliability have been factored into the subsystem design in terms of operation of normal and redundant modes, decision-making, and switchover to alternate modes.

Reference 1. "Building Block Approach in Forming a Multifunction Communications System" by M.B. Schulman, GE-MSVD. Advanced Instrumentation and Communication Memo No. 31, July 27, 1960.

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Where applicable, numerical estimates have been used to pinpoint problem areas and to compare competitive design elements.

In summary, the preliminary design effort has established not only a functional capability from the standpoint of performance, but also a reasonable assurance of a design of high reliability potential. In support of this contention, the following sections will consider some of the more pertinent aspects of reliability in context with the APOLLO system.

2.6.1 Numerical Analysis

In this discussion, it is assumed that equipment has been suitably "burned-in" before flight so that a constant failure rate will apply over the length of the missions, and only chance failures can occur at random. This leads to the selection of the simple exponential distribution of probability:

$$P_{(t)} = e^{-\lambda t} \quad (1)$$

This is the probability that there will be no failure of a component, subsystem, or system element, subject to a constant failure rate λ , within a given time interval (t). A procedure which could be used to provide numerical estimates is outlined in Appendix SC-D. This includes a description of the subsystem from the viewpoint of operational component dependencies, operating periods, and areas of redundancy. A probabilistic model is obtained and failure rates are combined to form a single equivalent failure rate and total reliability for the entire subsystem over its various operating modes. Although such estimates in preliminary design are predictions and can have only slight quantitative basis in fact, they can be of merit in pinpointing low-reliability problem areas for design improvement and provide initial criteria for trade-off considerations with performance and development time and cost. Some subsystem estimates are given in Appendix SC-E.

In the analysis of subsystem reliability, consideration must be given to the role of the crew. This subject is covered in VOLUME V with regard to responsibilities and task performance. In addition, the subsystem descriptions in the volumes of this report include the crew dependencies. A subsystem reliability analysis does not necessarily have to include, initially, man's contribution to reliability by means of failure sensing, decision making, and backup in the form of switching or maintenance; this can be factored-in later. The role of man as a series initiator or operator can be treated independently from a subsystem reliability estimate where the equipment estimate is not compromised (i.e. where

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estimates are used for comparing competing components with similar dependencies on manual operation).

It can be assumed, for the present, that man is available when required to perform the necessary functions that have been assigned. This assumption can be justified on the basis that the crew can be preflight-conditioned to a high operational level of proficiency and that a three-fold redundancy is available in the crew. The degradation of crew performance under environmental stress has yet to be determined. Data on this will become available from the Mercury program and during the course of the APOLLO research and development program.

2.6.2 System Reliability

System composition will vary from mission phase to mission phase. Evaluation of per-phase reliability must include the consideration of not only the per-phase subsystem reliabilities but also the dependencies of the system on the subsystem elements. It is at the system level that it is first convenient to consider separately the reliability related to mission success and that related to crew safety. The statement of operation and dependencies which must be made (in a manner similar to that for the subsystems) could include the double definition for success - one for the mission and one for crew survival. At the per-phase system level, crew survival is defined as a function of environmental stresses and mission success is defined in terms of crew survival and trajectory considerations.

The probability distribution of the system with time will have a high degree of variation over the total mission, however, since the system phase profile is apportioned as the least-common-denominator of the subsystem modes. The system reliability will have to be considered over the emergency modes as well as the normal mode in order to evaluate the potential for crew survival.

Analysis of the per-phase reliability can be simplified in an initial investigation by: (1) minimizing the variation during the mission phases by including only those major phases where basic subsystem dependencies prevail, and (2) conservatism in any estimates by assuming all pertinent subsystem functions are completely necessary for success of the phase. Tables II-6-Ia and b are simplified checklists of the type recommended for use in the normal and emergency mission modes. A further simplification is implied, namely, there is no objective mission success if an emergency mode is resorted to or

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TABLE II-6-I SIMPLIFIED CHECKLIST OF MISSION PHASE DEPENDENCIES

| (a) Normal Mission | Electrical Power | Environmental Control Function | Navigation Guid. & Cont. | Propulsion | Communications | Landing | Recovery Aids |
|---|------------------|--------------------------------|--------------------------|------------|----------------|---------|---------------|
| Powered Flight | (X) | (X) | | | | | |
| Midcourse-to-Moon | (X) | (X) | (X) | (X) | (X) | | |
| Lunar Orbit | (X) | (X) | (X) | (X) | (X) | | |
| Midcourse-to-Earth | (X) | (X) | (X) | (X) | (X) | | |
| Re-entry | (X) | (X) | (X) | | | | |
| Landing | (X) | (X) | | | | (X) | |
| Recovery | (X) | | | | | | (X) |
| (b) Abort/Emergency Modes | | | | | | | |
| Abort/Off-the-pad and first booster stage | X | X | | | | X | X |
| Abort/Second booster stage | X | X | X | | | X | X |
| Abort/Third booster stage | X | X | X | | | X | X |
| Emergency Return From Midcourse-to-Moon | X | X | X | | | X | X |
| Shorten Lunar Orbit | X | X | X | | | X | X |

X Applicable to both crew survival and mission success

X Applicable to crew survival only

the crew does not survive. It should be noted that the subsystem for any one function will not be the same through all phases. In fact, a function failure conceivably can be the reason for an emergency return and yet, while operating in an emergency mode, be a determinant for the success of the return. An example of this is emergency return due to space power system failure. During the return, the requirement for electrical power would be satisfied for the critical functions by the auxiliary or battery supply.

Table II-6-Ia gives a gross indication of the strictness of the requirements for a completely successful mission. If each of the 25 elements listed as required for the complete mission had a reliability of 0.995, for example, the total probability for success

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of the objective mission would be 0.882 (i.e. $0.995^{-(25)}$). (This is without the Saturn booster factored-in.) However, modal analysis of the type presented in Appendix SC-C with considerations of actual dependencies would realistically raise this estimate since not all subsystem functions are 100 percent essential, even though some dependency is indicated.

2.6.3 Mission Accomplishment and Crew Survival

The probability of mission accomplishment could be expressed as a distribution of probability of successfully attaining a point in time of the objective mission. An illustration of this is given in Figure II-6-1a. The distribution has the appearance of the successive per-phase system probabilities in cascade and can be expressed as

$$P(t) = e^{-\lambda_k(t - t_{k-1})} e^{-\sum_{i=1}^{k-1} \lambda_i(t_i - t_{i-1})} \quad (2)$$

with t occurring in the k th phase.

In order to consider the effect of alternative missions, however, it is more convenient to consider the probability of completion of the mission ($P_R(t)$). This is illustrated for the objective mission in Figure II-6-1b. It is derived from Equation (2), as follows:

$$P(t_n) = P(t) \cdot P_R(t) = e^{-\sum_{i=1}^N \lambda_i(t_i - t_{i-1})}, \quad (3)$$

$$\text{and:} \quad P_R(t) = \frac{P(t_n)}{P(t)} = e^{-\lambda_k(t_{k-1} - t)} e^{-\sum_{i=k}^N \lambda_i(t_i - t_{i-1})} \quad (4)$$

The goal of 90 percent for mission accomplishment discussed earlier, is indicated on Figure II-6-1b as the initial point of the mission.

The effect of the sequential redundancy presented by the emergency modes is to increase the probability of crew survival. The application of a particular emergency mode in the event of a failure is limited over a range of time which could be within a single, or span several, normal mission phases.

Now, assume that all $P_R(t)$ for all times working backward from recovery to time t (occurring in the k th phase) have been determined. Assume that the determination of

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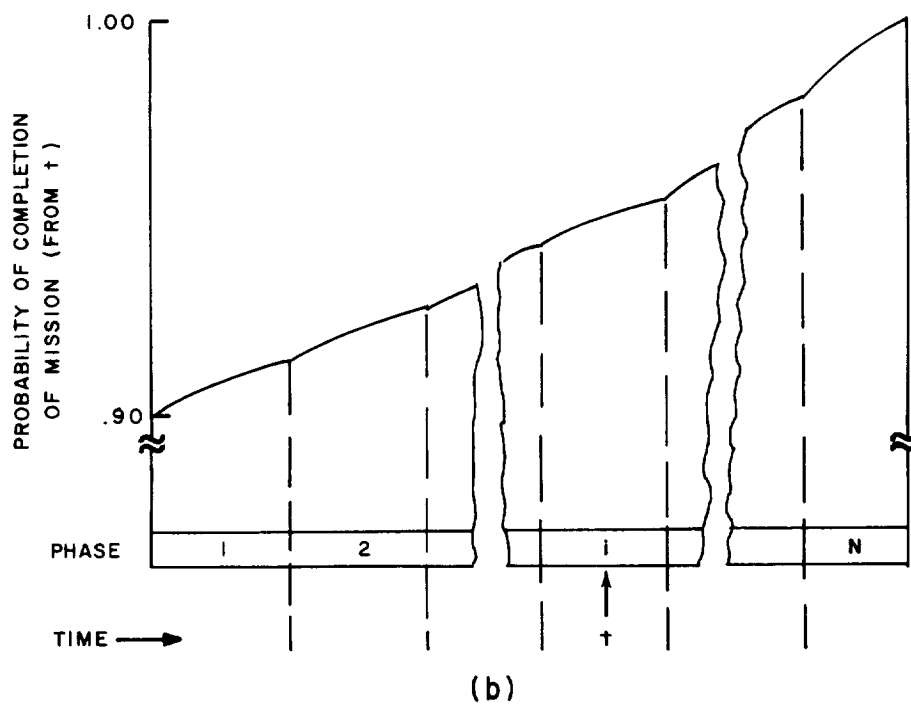
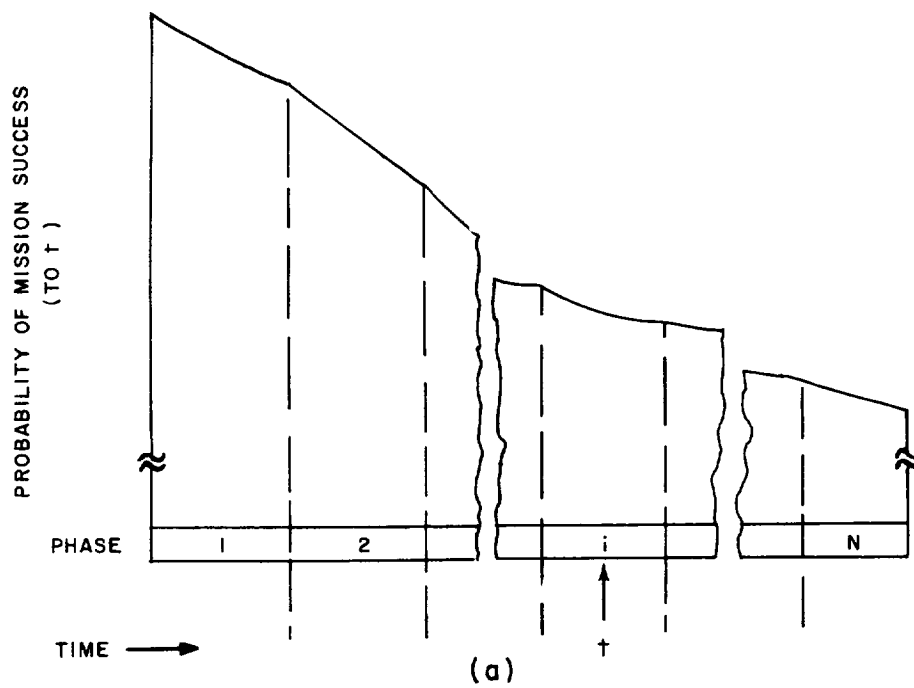


Figure II-6-1. Normal and return probability of mission success

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these $P_R(t)$ factored-in the sequential redundancies available in the form of emergency modes, both subsystem and system. The λ_i for all times and phases after t_k will be equivalents based on not only system-interacting subsystems but also the redundant emergency modes where they are applicable. See Figure II-6-2.

Under the above conditions, equation (4) can be said to represent the probability of return by way of the normal mission from time t and all emergency alternatives after the k_{th} phase. The probability of safe return from time t ($P_{SR}(t)$) would then be the probability of an available emergency alternative mode (A_k) in sequential redundancy to: the k th normal mission phase in cascade with $P_{SR}(t_k)$.

$$\text{Then: } P_{SR}(t) = P_R(t) + [1 - P_R(t)] P_{SR}(t_{k,o}) \quad (5)$$

The determination of $P_{SR}(t_{k,o})$ would include further complications where second-order alternatives of A_k are available. Generally, however,

$$P_{SR}(t_{k,o}) = e^{-\sum_{i=1}^M \lambda_{k,i} (t_{k,i} - t_{k,i-1})} \quad (6)$$

In many cases, the time duration of the alternate mission will be a function of the time of application ($t_{k,o} = t$). This would mainly be expressed in the first phase or two of the alternate mission ($A_{k,1}; A_{k,2}$). (An example of this is emergency return from the mid-course-to-Moon phase. Here, the application of maneuver thrust and return to re-entry point would be time-variable depending upon point of return decision). For this general case we have:

$$\begin{aligned} t_{k,o} &= t_1 \\ (t_{k,1} - t_{k,o}) &= (t_{k,1} - t) = f_1(t - t_{k-1}), \\ (t_{k,2} - t_{k,1}) &= f_2(t - t_{k-1}), \quad \text{etc.} \end{aligned} \quad (7)$$

From this development, we can see that probability of mission success from any point in time of the normal mission can be obtained from equation (5) by factoring in equations 4, 6, and 7. We could settle for less by considering only end points of normal mission

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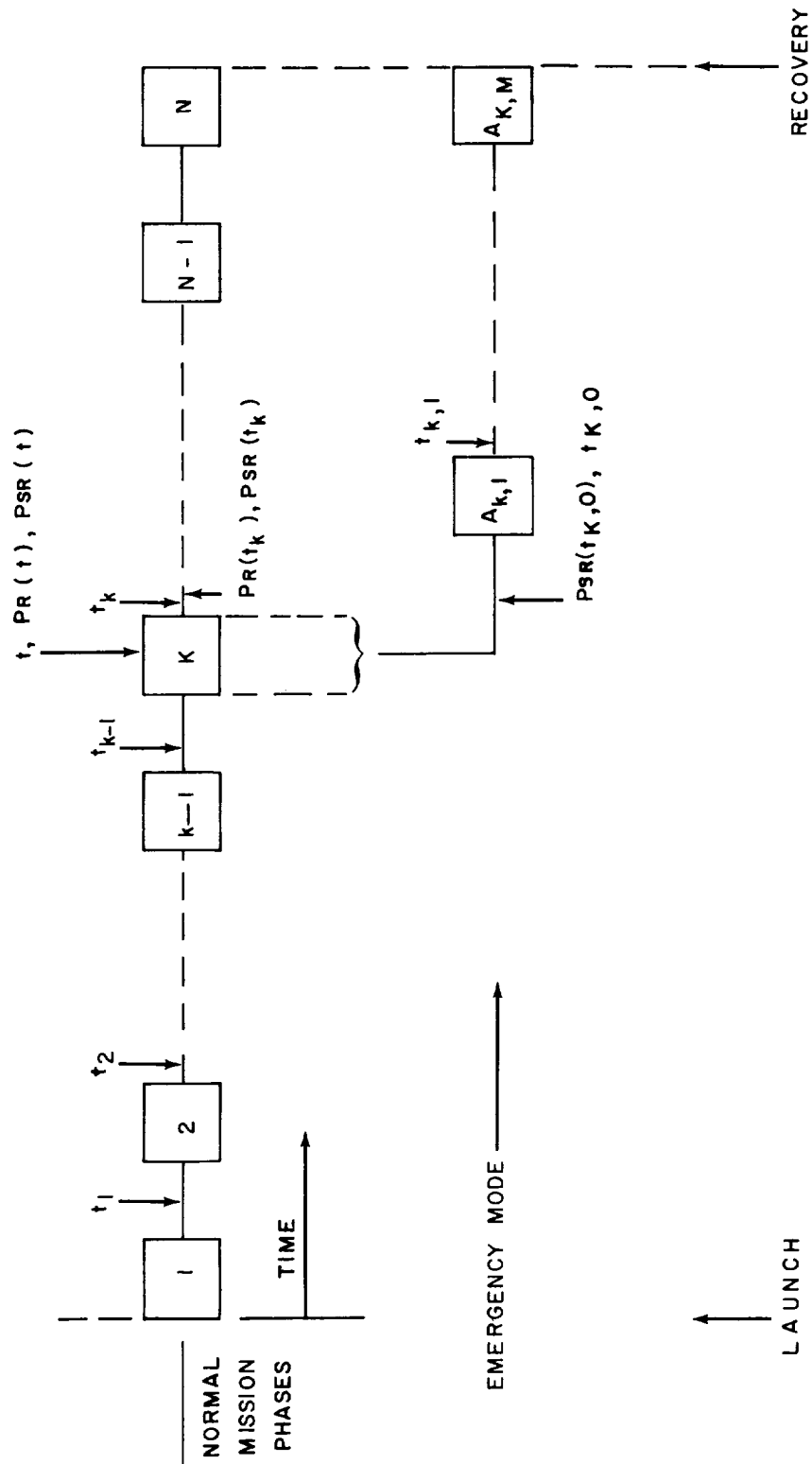


Figure II-6-2. Modal structure relating normal and emergency mission modes

phases. This approach is not in great error if the mission is phased in short enough segments. By using the shorter segments and end-points of application, the time duration of alternate modes can be fixed with the applicable t_i . With these simplifications, we have:

$$\begin{aligned}
 &P_R(t_{k-1}) = \prod_{i=k}^N P_i \\
 \text{and} \quad &P_{SR}(t_{k,0}) = \prod_{i=1}^M P_{k,i} \\
 \text{where: } &\prod = \text{product of.} \\
 &P_i = e^{-\lambda_i(t_i - t_{i-1})} \\
 &P_{k,i} = e^{-\lambda_{k,i}(t_{k,i} - t_{k,i-1})}
 \end{aligned} \tag{8}$$

Equations (8) and (5) in conjunction with a modal structure of normal and alternate mission modes could be used to develop the probability of crew survival.

2.6.4 Man's Contribution to Reliability

The complete delineation of the role of the crew in the APOLLO mission will be the result of a continuous iterative process involving considerations of system performance and system reliability with task apportionment between the crew and equipment. At the preliminary design stage, this iterative process has been entered into by means of a series of approximations related to the nature and operational aspects of the equipment and to the anticipated crew task structure. Previous discussion of the methodology of analysis in this section has omitted specific determination of the human factor in subsystems or system operation. This is done by intent in order, first of all, to evaluate equipment designs which are, in a coarse sense, independent of the man. This independence is rationalized on the basis that the crew is preflight-conditioned to a high operational level of proficiency and a three-fold redundancy is available in the crew. Furthermore, the operating task assignments have been kept to a reasonable level without assignment of mundane automatic reactions to stimuli which a simple mechanized link could serve. The attempt has been made to assign mode transition and in-line operational tasks to the crew where it was decided a favorable balance between performance and reliability would be obtained. These decisions have been decidedly weighted towards exploitation of the human facility for sensing and decision making and with regard for the environmental profile to which the crew will be subjected.

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In addition to parts selection, redundant design is the key to the success anticipated of the preliminary design presented in this report. Most instances of working redundancy (i. e. continuous parallel operation) are automatic functions, whereas most sequential redundancy applications are dependent on crew sensing of a failure and application of judgement. These latter redundancies range from the overall system level, in transition to emergency modes such as abort; through the subsystem level, with switching between available modes; to the component level, with maintenance (including replacement).

In the preliminary design, man's contribution to reliability is factored in where the equipment includes a switch, or the like, for transition between operating modes in the event of a failure. For the reasons given above, no significant loss of rigor in a reliability estimate would be incurred through inability to assign a failure rate value to the man as a component. As a matter of fact, offsetting this, neither have the maintenance capabilities of the crew yet been fully exploited in a specific manner.

As mentioned before, man's most important contribution to reliability is in the role of sensor-judge-switcher for sequentially redundant components, subsystems and mission modes. The first of these is maintenance where replacement is involved. The maintenance task is discussed from the human factors viewpoint in Volume V of this report. This discussion considers the maintenance functions within the special constraints of the closed-system APOLLO vehicle. Included are considerations of timing, priority assignment, and the possible remedial steps.

The remedial methods which could be followed in a maintenance function are:

Repair - return component to an acceptable operating condition by restoring original parts to "as new" condition (e. g., tighten, seal)

Replace - provide a substitute portion of mode of operation through:

- (1) use of module or component from "stock"
- (2) switching to equivalent substitute
- (3) switching to functionally equivalent alternative mode
- (4) "pirating" substitute part from a lower-priority subsystem

Delete - eliminate an operation or function to prevent impending malfunction.

Degrade - operate component at less than rated level, and/or intermittently.

Prevent - detect incipient malfunction via calibration or operational checks and prevent by one of the above methods, as appropriate.

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Each of these methods has potential application for same subsystem, for same phase, of the APOLLO mission. Replacement is a particularly rewarding area, especially where the total design has a significant degree of duplication or modularity between subsystems or components which are basically redundant or differ by a wide margin of priority. An approach to development of effective spares in the design has been discussed previously in Section 2.4 of this chapter.

2.7 FAILURE EFFECTS

An objective in the preliminary design has been to eliminate the possibility of failure within physical and performance limitations. Since a great number of safety precautions have been included in the preliminary design considerations, there are a very large number of detailed procedures which could be followed to ensure survival of the crew. Therefore, a more reasonable, although negative measure of the effectiveness of the design is the consideration of the number, and probability of occurrence, of the possible methods by which the crew could be killed (one or more) or the mission objectives defeated. Crew survival is the critical factor, given the objective mission flight profile as a requirement. The probability of occurrence of a large number of simultaneous, independent failures which could cause a crew-kill are infinitesimal. The evaluation can be confined to the greater likelihood of those single and double failures which could result in a kill.

The probabilities of single and double failures of passive (including structure) and active elements which could cause a crew-kill have a relationship as shown in Figure II-7-1. Also noted is the relative effect of these failure types. The relation between effects is primarily due to the generally more disastrous consequences of a failure in a passive element caused, in design, by heavier dependency on the greater material strength of the passive element. As a result of these relationships, the probability of a crew-kill drops to a consistently low level. The value of this level does not admit of any qualitative significance for internally caused failures, due to the redundancies and emergency modes built into the preliminary design.

Certain externally caused failures, either alone or in combination with internal causes, could lead to some significant probability of crew-kill. These external causes are:

- Booster failure
- Excessive radiation
- Meteoroid penetration

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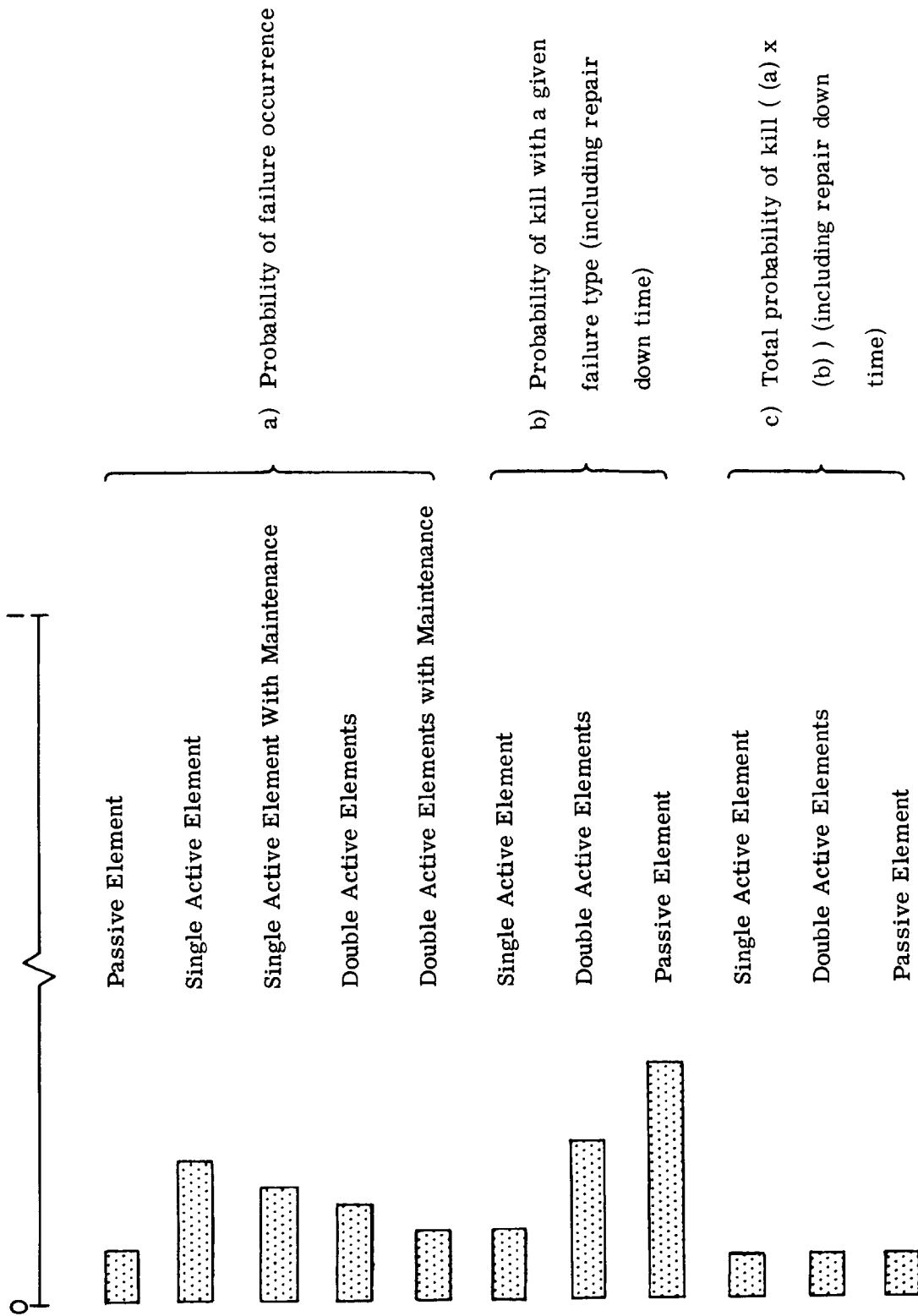


Figure II-7-1. Relationship between internal failure types relative to crew kill.

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It should be noted that this short list does not include those adverse conditions for which there is adequate protection for the crew in view of the risk. As an example, excessive vibration during boost could conceivably cause leaks in the vehicle seals. It is extremely unlikely that these leaks would be large enough to cause a significant loss of command module pressure. If this did occur, however, the secondary pressure system could be used. Furthermore, since it is extremely unlikely that any leaks would occur or increase in size after the launch phase due to launch phase vibration, the secondary pressure system would be required as the primary protection only over the short time required for an emergency return immediately after injection.

From loxing through powered boost to injection, there is a time-variable probability of booster failure. This failure could be an explosion, failure to ignite, loss of thrust, failure to separate, booster guidance or control failure, or others. (These are discussed in Volume II). Associated with booster failure is the requirement for advance warning, especially in the case of imminent explosion. Also required is the firing of one or more separation rockets and certain numbers of abort rockets. The number of abort rockets is based on pessimistic estimates of potential magnitude of the booster explosion, an overpressure of 10 psi (in spite of structural design integrity of up to 30 psi) and a warning time of 2 seconds. Based on these considerations, Table II-7-1 gives a simplified presentation of the rocket requirements for safe return in the event abort becomes necessary.

TABLE II-7-1. MINIMUM ROCKET REQUIREMENTS FOR ABORT

| | SEPARATION ROCKETS | ABORT ROCKETS | |
|--------------------------|-----------------------|---|---|
| | | Without Booster Shutdown | With Booster Shutdown |
| From pad to high q | 2 of 4 | 7 of 8 | 7 of 8 |
| At 1st stage burnout | 1 of 4 | 3 of 8 | 2 of 8 |
| At 2nd stage ignition | 1 of 4 | 3 of 4 | 2 of 4 |
| At 2nd stage burnout | 1 of 4 | 3 of 4 | 1 of 4 |
| At 3rd stage ignition | none | 1 of 2 | 1 of 2 |
| At 3rd stage burnout | none | 2 of 2 or 1 of 2 with on-board pro- pulsion | 1 of 2 without on-board pro- pulsion |

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Radiation level excesses are a measure of the preflight estimates of solar activity. As discussed in Volume V, Section 2.4, the probability of high radiation intensity due to solar activity will depend on the year of launch. Although the probability of exceeding the dosage level of 5 rem will range from less than 0.01 to as high as 0.125 (depending on year of launch), the kill probability is much less. Provided a quick emergency return is made and therapeutic methods applied immediately, dosages up to 750-1000 rem can be considered as non-killers. The probabilities of these are considerably less than 0.005.

The effect of meteoroid penetration is much more complex than the other factors. Penetration of the command module, mission module, propulsion section, or heat shield could lead to loss of part or all of the crew. The significant effects of penetrations of the command or mission module are loss of pressure and possible loss of a crewman or a critical portion of a subsystem required for safe return. Penetrations of the propulsion section could cause trouble where a combination of working parts of the system was penetrated, fuel loss occurs, or combustion or explosion occurs. Penetration of the heat shield could conceivably compromise the protection capability of the shield and result in disastrous re-entry. However, the critical forebody shield would have to be penetrated after the propulsion section.

In Table II-7-II, a simplified presentation is given for the various alternative effects of meteoroid penetrations. An order of magnitude approximation is given for the various alternatives. The estimates for pressure loss are pessimistic in that they are based on an assumed fixed procedure in the event of a penetration: All three crewman immediately enter the cocoons, with one donning the pressure suit subsequently.

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TABLE II-7-II.

ESTIMATED PROBABILITIES OF CREW-KILL DUE TO METEOROID PENETRATION

| Meteoroid Penetration of | Subsequent Effect | Estimated probability of total effect relative to crew-kill |
|--------------------------|---|---|
| Mission Module | Direct impact of crewman | < 0.001 |
| | Excessive loss of pressure before escape into cocoons | < 0.001 |
| | Penetration of equipment critical to return (no repair possible) | < 0.0001 |
| Command Module | Direct impact of crewman | < 0.0001 |
| | Excessive loss of pressure before escape into cocoons | < 0.0002 |
| | Penetrations of one or more cocoons with excessive pressure loss | < 0.0002 |
| | Penetration of equipment critical to return (no repair possible) | < 0.0001 |
| Propulsion Module | Excessive pitting of forebody heat shield protection | < 0.0001 |
| | Penetration of combination of working parts of system required for return | < 0.0005 |
| | Loss of fuel required for safe return | < 0.0001 |
| | Penetration of oxygen tank. Prolonged and excessive combustion of aluminum | < 0.0001 |
| | Penetration of oxygen and hydrogen tanks to cause disastrous explosion in spite of high venting speed | < 0.00001 |

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CHAPTER III

ABORT

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III. ABORT

1.0 Summary

1.1 SELECTED ABORT SYSTEM

The system evolved for use with the selected APOLLO configuration, D-2, provides an abort capability through all stages of boost propulsion and an emergency return capability for both orbital and lunar missions as required. The system, activated either manually or by automatic means, utilizes eight solid propellant abort rockets to provide the required separation from the Saturn vehicle during boost flight. The effect of abort rocket weight upon the APOLLO mission payload capability is minimized by jettisoning selected rockets at various points during the ascent trajectory. Additional solid propellant rockets are utilized to provide positive separation of the forward and aft sections of the APOLLO space vehicle from the self-contained re-entry vehicle at times which have been optimized for each specific phase of the mission in which abort may occur.

For aborts off the launch pad through Saturn stage S-II burnout, no additional thrust, over that of the abort rockets, is required. The resulting ballistic flight path can be followed without exceeding the acceleration tolerance of the crew and impacting within 1400 nmi of the launch site. For all phases of the mission beyond Saturn stage S-II burnout, the APOLLO midcourse and lunar orbit propulsion systems (on-board propulsion) will be utilized to provide the velocity vector increment required to return the re-entry vehicle to a suitable recovery location.

The nature of the primary APOLLO mission and the requirement for a versatile vehicle which can be made adaptable to other missions predicated the selection of a preliminary design which, upon initial inspection, seems to require a relatively complex abort sequence. However, every effort has been made to reduce complexity and increase the reliability of the over-all system. An example of this design philosophy can be seen in the abort parachute deployment system which, for a majority of the boost profile, uses the same operating sequence as for the normal orbital or cislunar mission. A further example, as can be seen in the next section, is the utilization of the Saturn C-2 staging sequence to initiate changes in the abort sequence, thereby eliminating the necessity for additional programming.

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1.2 ABORT SEQUENCE

The APOLLO mission encompasses a spectrum of flight conditions ranging from the static conditions on the launch pad through boost flight to orbital and escape velocities and subsequent cislunar flight. The abort and emergency return sequence and modes of operation must correspondingly be fitted to the particular mission phase at the time of required action. Therefore, the abort system described in this report utilizes a pre-determined sequence of events which, in turn, is dependent on the mission phase. The major events, which change or modify the abort sequences, are listed in Table III-1-I. The sequences for each mission phase are listed in Tables III-1-II through III-1-VII. Detailed descriptions of each event are included in Section 2.0, Chapter III, Systems Operation. The sequence for either launch pad or max q abort is shown, as an example, in Figure III-1-1.

TABLE III-1-I
MAJOR EVENTS AFFECTING PORTIONS OF THE APOLLO ABORT SEQUENCE

| TIME FROM LAUNCH | EVENT |
|------------------|--|
| 80 sec. | Timer arms baroswitch for primary recovery system initiation. |
| 98.2 sec. | Saturn stage S-I burnout/S-II ignition. Drop 4 abort rockets. |
| 284.9 sec. | Saturn stage S-II burnout/S-IV ignition. Drop 2 abort rockets, change separation point so that abort vehicle now has same interface as space vehicle (on-board propulsion remains with abort vehicle). Re-entry vehicle now remains within space vehicle until re-entry at 400,000 ft. Recovery system now actuated by normal re-entry sequence. |
| 774.68 sec. | Saturn stage S-IV burnout and separation. Drop remaining abort rockets and separation rockets. |

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TABLE III-1-II
LAUNCH PAD ABORT SEQUENCE

| <u>TIME (Sec.)</u> | | <u>EVENT</u> | <u>REMARKS</u> |
|--------------------|-------------|---|---|
| From abort | From launch | | |
| -2.0 | -- | Minimum abort warning time | See Section 5.1 |
| 0 | -- | Abort rockets ignited | Shaped charge separation from APOLLO vehicle is followed immediately by ignition of 8 abort rockets. |
| 2.5 | -- | Abort rocket burnout | |
| 4.5 | -- | Re-entry vehicle separation | Solid rocket separation of forward (including mission module) and aft space vehicle structure sections. |
| 12.5 | -- | Initiate recovery sequence | Timer initiated at approximately apogee. |
| 19.7 | -- | First stage chute (First ribbon) fully open | See Section 5.5 for detailed chute sequence. |
| 31.0 | -- | Main chutes open in reefed condition. | |
| 37.5 | -- | Main chutes fully open | |
| 80.1 | -- | Impact | Impact velocity (no wind) 30.3 ft /sec. |

TABLE III-1-III
MAX. 'q' ABORT SEQUENCE

| <u>TIME (Sec.)</u> | | <u>EVENT</u> | <u>REMARKS</u> |
|--------------------|--------------------|---|---|
| <u>From abort</u> | <u>From Launch</u> | | |
| 0 | 72.0 | Abort rockets ignited | Shaped charge separation from APOLLO vehicle is followed immediately by ignition of 8 short rockets. |
| 2.5 | 74.5 | Abort rocket burnout | |
| 4.5 | 76.5 | Re-entry vehicle separation | Solid rocket separation of forward (including mission module) and aft space vehicle structure sections. |
| 12.5 | 84.5 | Initiate recovery sequence | Timer initiated near apogee. |
| 19.7 | 91.7 | First stage chute (First ribbon) fully open | See Section 5.5 for detailed chute sequence. |
| 31.0 | 103.0 | Main chutes open in reefed condition | |
| 37.5 | 109.5 | Main chutes fully open | |
| 1015.0 | 1087.0 | Impact | Impact velocity (no wind) 30.3 ft /sec. |

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TABLE III-1-IV

ABORT SEQUENCE AT SATURN STAGE S-I BURNOUT AND STAGE S-II IGNITION

| <u>TIME (Sec.)</u> | | <u>EVENT</u> | <u>REMARKS</u> |
|--------------------|--------------------|-----------------------------|--|
| <u>From Abort</u> | <u>From Launch</u> | | |
| 0 | 98.2 | Abort rockets ignited | Shaped charge separation from APOLLO vehicle is followed immediately by ignition of abort rockets. |
| 2.5 | 100.7 | Abort rocket burnout | |
| 4.5 | 102.7 | Re-entry vehicle separation | Solid rocket separation of forward and aft space vehicle structure sections. |
| 165.6* | 263.8* | Initiate recovery sequence | Baroswitch initiated at 25,000 ft. |
| 157.7** | 255.9** | | |
| 796a* | 894a* | Impact | Range from launch pad: 32.7 nmi |
| 788a** | 886a** | | |

*At Stage S-I burnout using 8 abort rockets

**At Stage S-II ignition using 4 abort rockets

a - approximate

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TABLE III-1-V
ABORT SEQUENCE AT SATURN STAGE S-II BURNOUT

| <u>TIME (Sec.)</u> | | <u>EVENT</u> | <u>REMARKS</u> |
|--------------------|--------------------|-----------------------------|---|
| <u>From abort</u> | <u>From launch</u> | | |
| 0 | 284.9 | Abort rockets ignited | Shaped charge separation from APOLLO vehicle is followed immediately by ignition of 4 abort rockets. |
| 2.5 | 287.4 | Abort rocket burnout | |
| 4.5 | 289.4 | Re-entry vehicle separation | Solid rocket separation of forward and aft space vehicle structure sections. |
| 498.5 | 783.4 | Initiate recovery sequence | Baroswitch initiated at 25,000 ft. Max g experienced in ballistic re-entry is 13.38 at 122,500 ft. (t = 362.5). |
| 1128.5a | 1413.4a | Impact | Range from launch pad: 1333 nmi |

a - approximate

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TABLE III-1-VI
ABORT/EMERGENCY RETURN SEQUENCE AT SATURN STAGE-IV IGNITION

| <u>TIME (Sec.)</u> | | <u>EVENT</u> | <u>REMARKS</u> |
|------------------------------|--------------------|---|--|
| <u>From Abort</u> | <u>From Launch</u> | <u>ABORT CASE</u> | |
| 0 | 284.9 | Abort rockets ignited | Shaped charge separation of APOLLO spacecraft from booster followed immediately by ignition of 2 abort rockets. |
| 2.5 | 287.4 | Abort rocket burnout | |
| 4.5 | 289.4 | Re-entry vehicle separation | Solid rocket separation of forward and aft space vehicle structure sections |
| 491.0 | 775.9 | Initiate recovery sequence | Baroswitch initiated at 25,000 ft. Max g experienced in re-entry is 13.57 at 120,800 ft. |
| 1121 | 1406 | Impact | Range from launch pad: 1065 nmi |
| <u>EMERGENCY RETURN CASE</u> | | | |
| 0 | 284.9 | Abort rockets ignited | Shaped charge separation of APOLLO spacecraft from booster followed immediately by ignition of 2 abort rockets. |
| 2.5 | 287.4 | Abort rocket burnout. On-board propulsion ignition. | Thrust application for $\Delta V \approx 4600$ fps. |
| 83.0 | 367.9 | Propulsion shut down | $V = 22692$ fps |
| 833 | 1118 | Re-entry vehicle separation | At 400,000 ft. Solid rocket separation of forward and aft space vehicle structure sections. Re-enter in equilibrium glide at $L/D = 0.6$ |
| 1220(a) | 1505(a) | Initiate recovery sequence | Baroswitch at 25,000 ft. |
| 1850(a) | 2135(a) | Impact | In vicinity of Ascension Island. |

(a) approximate

TABLE III-1-VII
EMERGENCY RETURN SEQUENCE AT ORBITAL VELOCITY

| <u>TIME (Sec.)</u> | | <u>EVENT</u> | <u>REMARKS</u> |
|--------------------|--------------------|--|---|
| <u>From abort</u> | <u>From launch</u> | | |
| 0 | 581 | Abort rockets ignited | Shaped charge separation of APOLLO spacecraft from booster followed immediately by ignition of 2 abort rockets. |
| 2.5 | 583.5 | Abort rocket burnout. Start re-orientation of spacecraft for retro | |
| 90 (a) | 671(a) | Start retro thrust using on-board propulsion system | Thrust application for $\Delta V \approx 500$ fps. |
| 100.5(a) | 681.5(a) | Propulsion shut down | $V = 25100$ fps |
| NA* | NA* | Re-entry vehicle separation | At 400,000 ft. solid rocket separation of forward and aft space vehicle structure sections. Re-entry made in equilibrium glide at $L/D = 0.6$ |
| NA* | NA* | Initiate recovery sequence | Baroswitch at 25,000 ft. |
| NA* | NA* | Impact | In vicinity of Ascension Island |

(a) approximate

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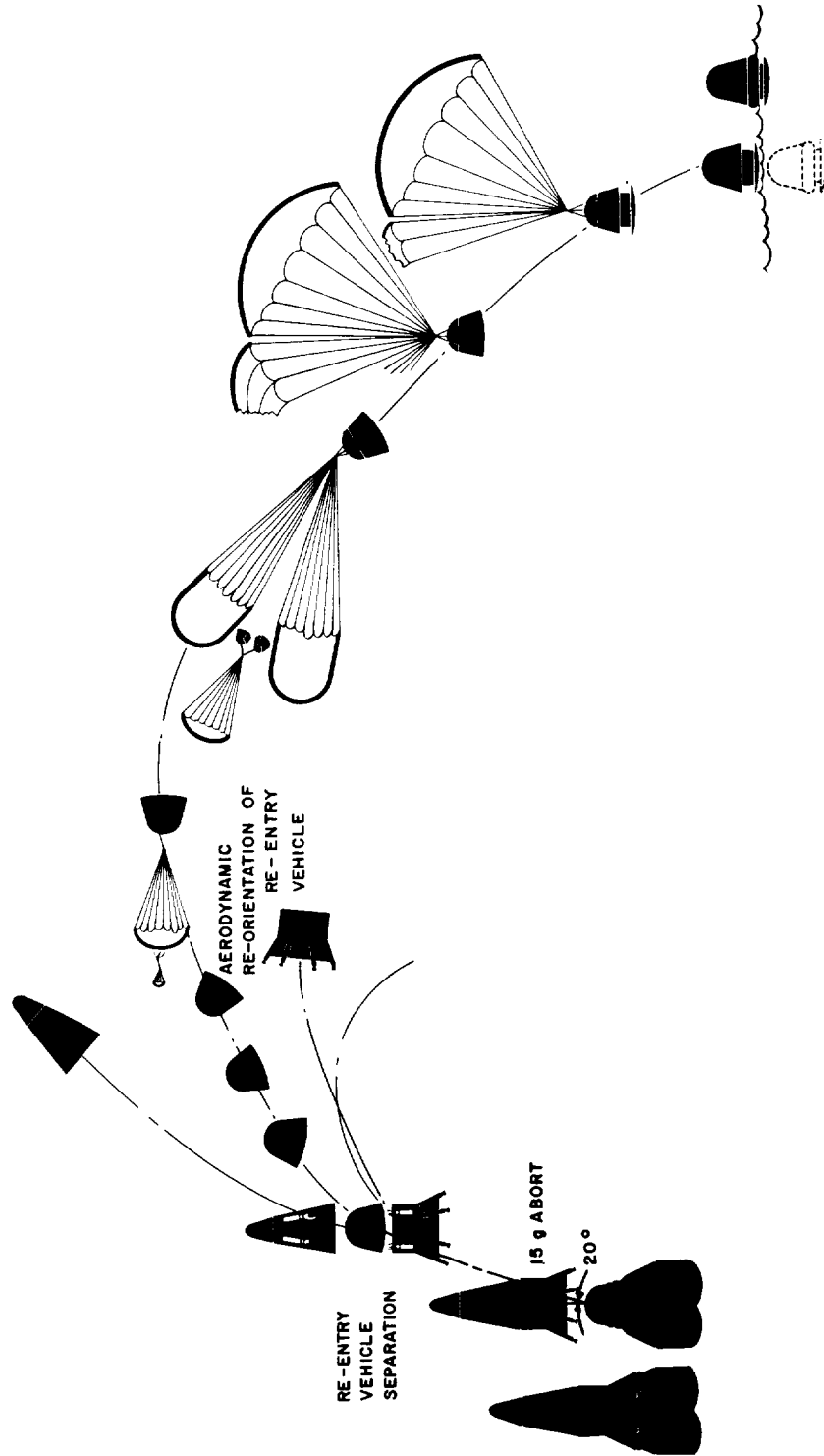


Figure III-1-1. Launch pad abort sequence

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1.3 R-3 CONFIGURATION ABORT SYSTEM

The basic difference between the abort systems for the selected D-2 configuration and the R-3 modified lenticular vehicle are the result of the complete enclosure of the D-2 re-entry vehicle within a protective aerodynamic shell and the ability of the modified lenticular vehicle to maneuver and effect a horizontal landing.

For conditions existing from launch thru Saturn stage S-II boost the re-entry vehicle is separated from the mission module, accelerated up and away from the booster, and maneuvered into a glide path from which a conventional landing can be made. If, after Saturn Stage S-II burnout and separation, the required escape velocity or flight path has not been achieved and mission abort is required, the propulsion module is used to provide the required velocity vector increment in a manner similar to that described for the D-2 configuration. The normal sequence for re-entry and recovery follows and a conventional landing is made.

The following sequence of events occur for launch pad abort of the R-3 configuration.
(Figure III-1-2)

- (a.) The fairing between the recovery vehicle and mission module is jettisoned.
- (b.) The recovery vehicle is separated from its attaching structure by gas operated disconnects and separation thrusters.
- (c.) Simultaneously with sequence (b), the six solid propellant abort rockets are ignited.
- (d.) Elevons move to neutral position within the first second of abort boost.
- (e.) The abort rockets are jettisoned after burnout.
- (f.) The windshield cover is immediately jettisoned to permit the pilot to orient himself for maneuvering.
- (g.) At the top of the escape trajectory the vehicle is rolled 180 degrees, from the inverted to the upright position, to allow a normal glide.
- (h.) The vehicle is maneuvered to a predetermined landing area.
- (i.) The parachute is deployed reefed after completion of the flareout and just prior to touchdown. This shortens the ground run and augments the longitudinal and directional stability.

1.4 SPECIFICATIONS OF THE APOLLO D-2 CONFIGURATION ABORT SYSTEM

| | |
|--------------------------------|---|
| Number of abort rockets | 8 |
| No. jettisoned at S-1 burnout | 4 |
| No. jettisoned at S-II burnout | 2 |

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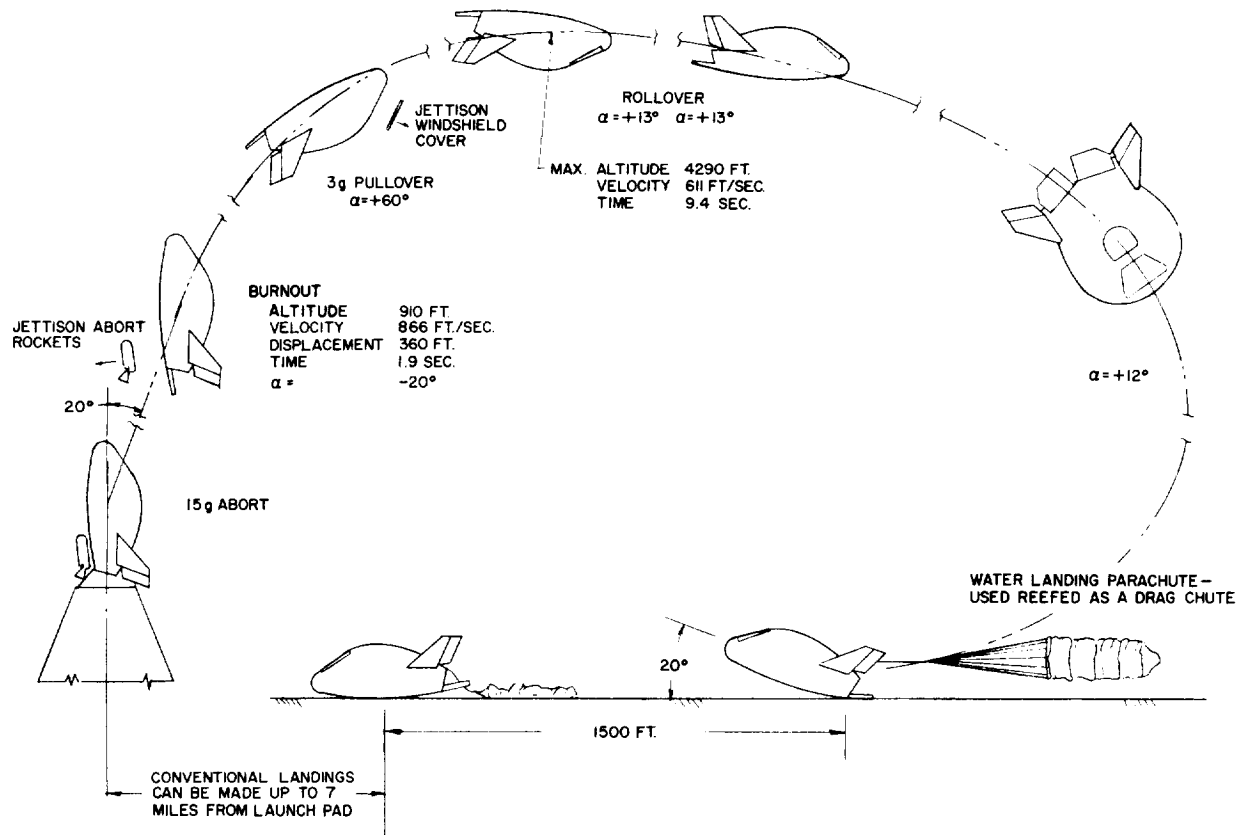


Figure III-1-2. R-3 configuration-launch pad abort

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| | |
|---|---------------------------------------|
| Abort rocket thrust (each) - initial | 19450 lb |
| - average | 17800 lb |
| Abort thrust inclination from vertical | |
| 8 rockets | 20.0 deg |
| 4 rockets | 21.2 deg |
| 2 rockets | 18.3 deg |
| Abort rocket burning time | 2.5 sec (nominal) |
| Aborted weight (including rockets) | |
| Launch pad | 9577 lb |
| After S-I separation | 8623 lb |
| After S-II separation | 7804 lb |
| After S-IV burnout | 7327 lb |
| Method of recovery sequence initiation | |
| Launch pad abort | timer set at 12.5 sec |
| Max q abort | timer set at 12.5 sec |
| After t = 80 | baroswitch with g-switch/timer backup |
| Recovery System | |
| 1st stage chute-Fist ribbon, 25 ft dia, $C_D A = 100$ sq ft | |
| Main chutes (3) - reefed, 19 ft dia. $C_D A = 845$ sq ft (total) | |
| Main chutes (3) - fully open, 53 ft dia. $C_D A = 4673$ sq ft (total) | |
| Separation Rockets | |
| No. for separation of forward space vehicle structure during Saturn S-I and S-II phases | 4 |
| Thrust - initial | 14790 lb |
| - average | 10240 lb |
| Burning time (nominal) | 1.12 sec |
| No. for separation of forward space vehicle structure after Saturn S-II burnout | 4 |
| Thrust - initial | 762 lb |
| - average | 642 lb |
| Burning Time (nominal) | 1.0 sec |
| No. for separation of aft space vehicle structure | 4 |
| Thrust - initial | 762 lb |
| - average | 642 lb |
| Burning time (nominal) | 1.0 sec |

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1.5 REFERENCE DATA

Reference data used in this study are indicated in the appropriate sections by footnotes. It should be noted, that prior to the mid-term presentation, very little information was available relative to the Saturn booster system and, in fact, some of the data referenced were not obtained until the closing weeks of the study.

The studies were therefore conducted using the data available at the time with new inputs being factored in as they became available. It was not possible however, to completely do over, or revise, all the work which had already been started or completed prior to the receipt of new information. There are, therefore, sections of this study which are based on reference data which have since been revised or modified. However, in no case are study results presented which, if revised to reflect the latest data inputs, would result in major differences in the study conclusions reached.

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2.0 Systems Operation

2.1 FUNCTIONAL OPERATION

The functional block diagram of the D-2 abort system is presented in Figure III-2-1. Functions are included for all phases of the mission. However, all are not necessarily operative during any specific phase. The following sections contain a description of each of the major subsystems indicated on this diagram. For reference, and to indicate relative locations of the major components, outline drawings of the APOLLO abort vehicle and spacecraft are included as Figures III-2-2 and III-2-3.

2.1.1 Crew Stations:

The prerogative for initiating an abort always rests with the crew. Of course, the exercise of this prerogative is fully dependent upon the information available to the crew relative to the necessity for abort and the decision time available which is a function of mission phase. Information relative to booster performance, trajectory and other flight parameters, maneuvering capability, cabin environment, and numerous other criteria will be presented by the cabin displays (see Volume V) in order to provide sufficient decision making capability for manned control of the vehicle.

A manually operated mode selector, for use in emergencies which occur after Saturn stage S-II burnout, determines the type of trajectory change to be programmed. Selection of the "Abort Mode" will provide the most expeditious return to earth without regard to landing site and should be utilized only in cases of extreme emergency. Selection of the "Emergency Return Mode" will provide, through the abort computer, a command to the guidance computer for the velocity vector correction necessary to effect a return to a predetermined landing site. If the vehicle has already reached superorbital velocity, the initial velocity vector connection will be that necessary to re-enter the atmosphere (400,000 ft) at the proper geographical coordinates for landing at the predetermined site. Additional corrections will be made as necessary. A detailed discussion of the guidance and control system is given in Volume III.

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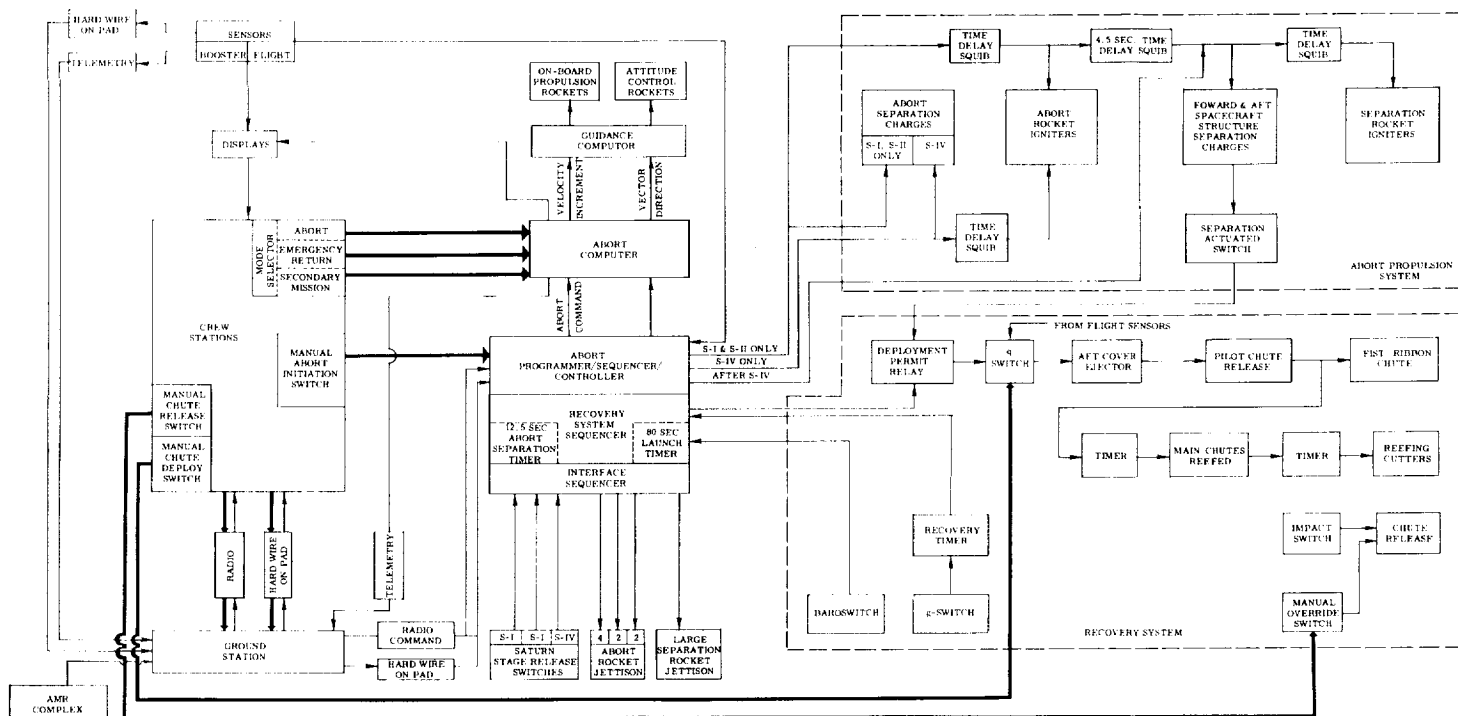


Figure III-2-1. APOLLO ABORT SYSTEM FUNCTIONAL BLOCK DIAGRAM

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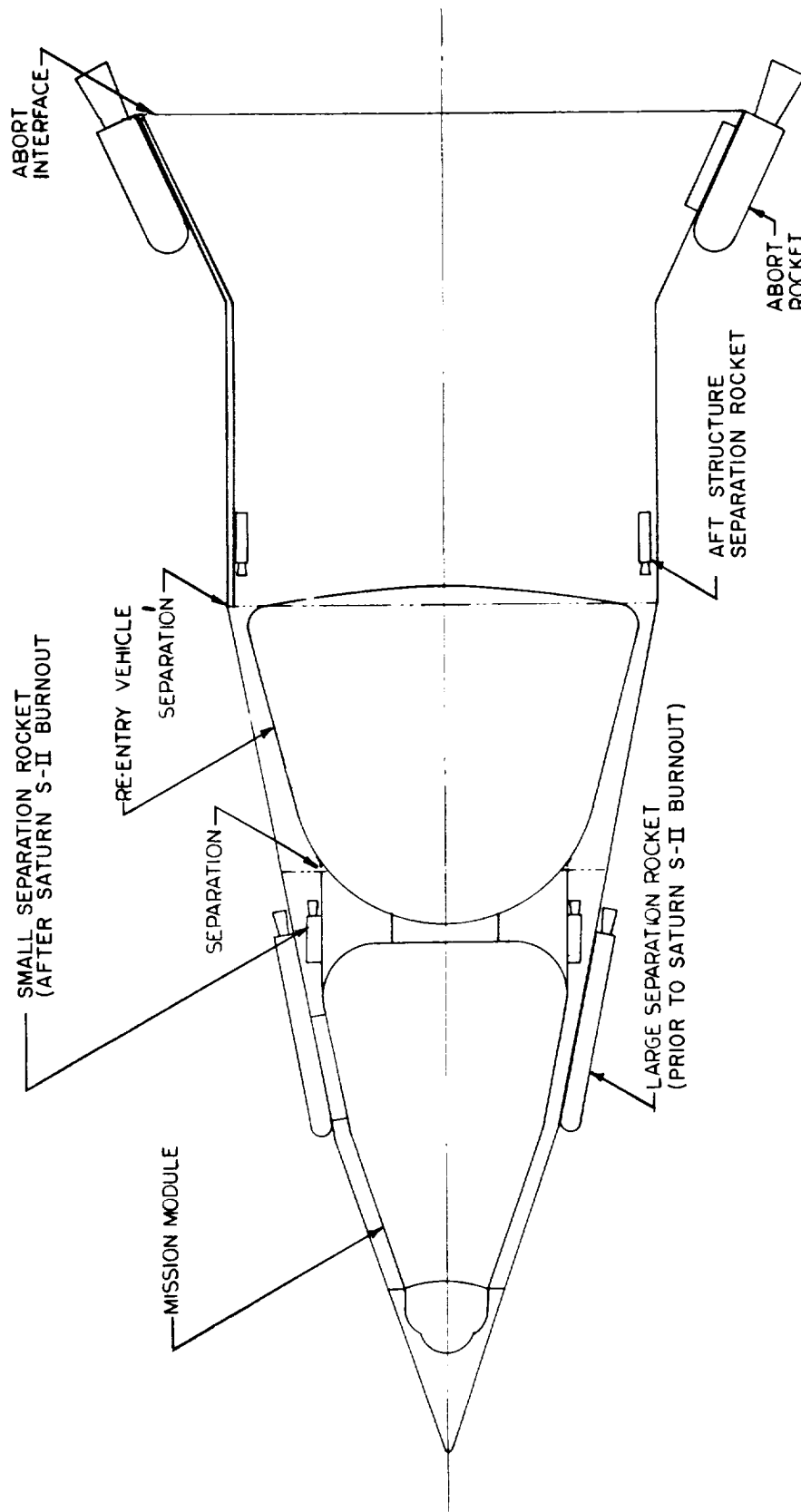


Figure III-2-2. APOLLO abort vehicle

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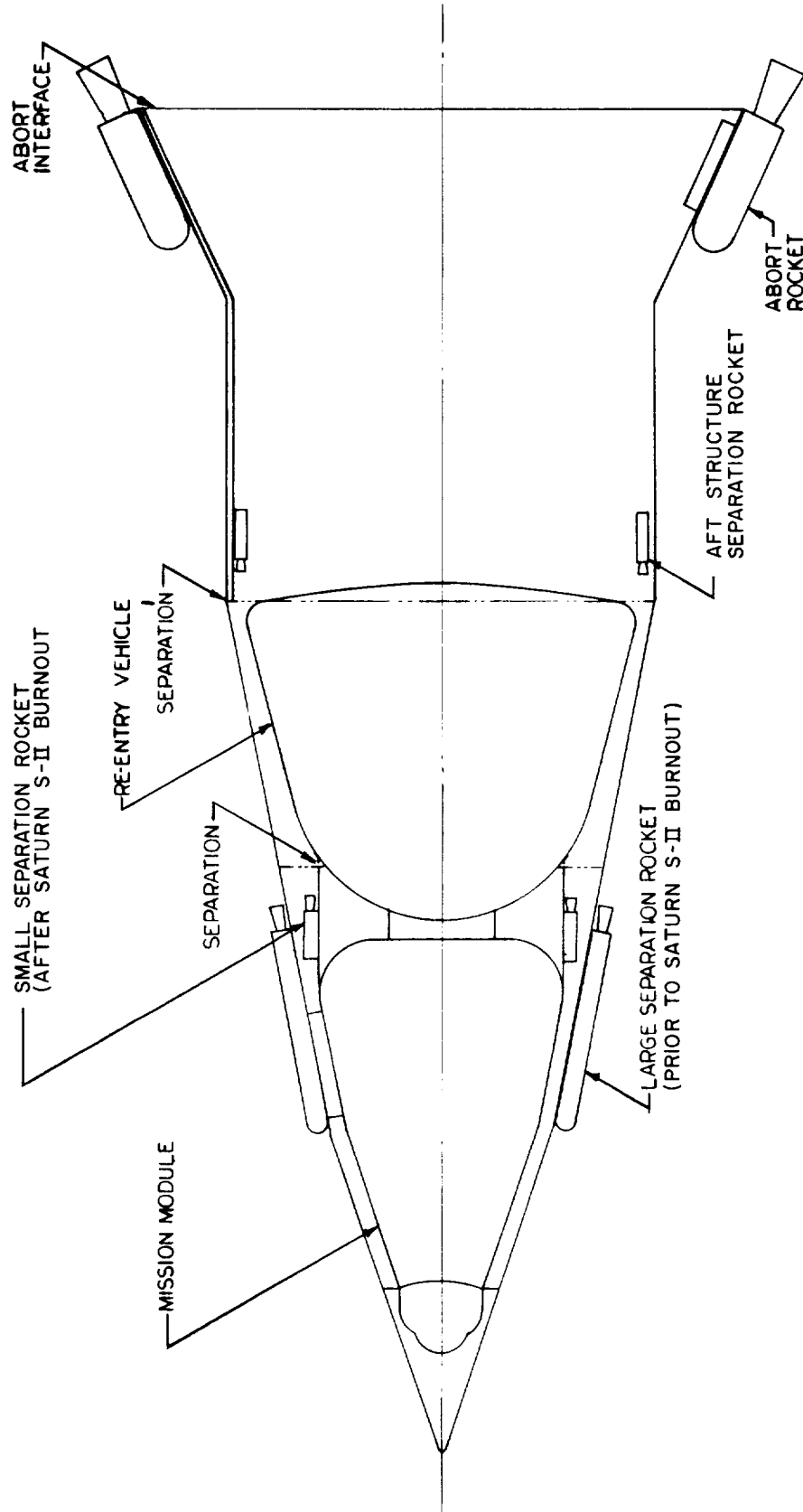


Figure III-2-3. APOLLO space vehicle (abort after Saturn S-II burnout)

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"Secondary Mission Mode" selection will provide for automatic vectoring; to a preselected secondary mission trajectory. This may, for example, be a transfer to an earth orbit during third stage boost or highly elliptical earth orbit during an early phase of cis-lunar flight.

Complimenting the mode selector is the Manual Abort Initiation Switch which, when activated, immediately starts the abort sequence.

Although the recovery system sequence is automatic to the extent that the parachute will be deployed at a minimum altitude of 25,000 ft, a Manual Chute Deployment Switch is provided to allow the crew to deploy the chute at a more desirable altitude or for emergency use. Similarly a Manual Chute Release Switch is provided in the event of failure of the impact actuated parachute release.

Supplementing the visual and audio displays will be voice communications between the crew and ground control centers by means of radio and, prior to launch, hard wire communications through the umbilical connection.

2.1.2 Abort Programmer/Sequencer

This unit contains the necessary mechanical controls and electronic circuitry necessary to function as the central abort controller. The booster and flight sensors will be monitored in order to provide a warning to the crew of impending malfunctions and, in the event a preset level is exceeded in a critical system, will initiate the automatic abort system. These limiting items include maximum booster tank pressure, minimum thrust level, minimum d. c. line voltage, maximum pitch rate, and many other criteria (see Section 3.0). The actual abort sequence is initiated in this unit with command signals to the abort propulsion and recovery systems. In the event of an automatic abort, the command is triggered by the self-contained automatic abort system utilizing intelligence provided by the sensors, as shown in Figure III-2-1. The manual abort command signal, whether initiated by the APOLLO crew or by a ground controller, will be fed to this unit and will initiate the sequence by overriding the automatic abort system.

In addition to acting as a central abort system programmer, this function will control the sequence of operation of the abort system, including the arming and disarming of certain subsystems as a function of mission phase. The interface section of the sequencer will receive Saturn stage separation signals to allow the abort and large separation rockets to be jettisoned in the order required.

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For the recovery system the sequencer selects, utilizing self contained timers, the proper sensor for initiation of the parachute deployment system. An 80-second timer, activated on Saturn lift-off, arms the 12.5-second abort timer and disarms the baroswitch and g-switch. 80 seconds after lift-off (approx. 45,000 ft. altitude) the 12.5 sec. timer is disarmed and the baroswitch becomes the primary chute deployment sensor with a g-switch activated timer as backup. Section 5.5 details the operation of this system.

2.1.3 Abort Computer

The abort computer continuously determines the total thrust application and vehicle orientation required to provide the velocity vector increment necessary for abort and emergency return after Saturn stage S-II burnout. Prior to S-II burnout the computer is not an active part of the abort system although it receives, through the abort programmer/sequencer, continuous intelligence from the booster and flight sensors in order to be available for instant use. The unit makes available to the crew, through the display console, a graphic representation of the predicted landing sites for abort or emergency return at any moment. This enables the crew, by means of the mode selector, to control the return trajectory to an extent dictated by the emergency conditions making return necessary.

During Saturn stage S-IV boost, continuous computation will be made of the velocity vector correction required for the minimum time return to the surface consistent with both vehicular and human factors limitations but without regard to the actual impact point. This location will be presented in the crew display as the abort impact area. In the event of an extreme emergency the crew will have the option of selecting this mode of return by placing the mode selector switch in the "Abort" position and providing an abort command thru the manual abort initiation switch. Also computed and displayed continuously, will be a prepared landing site, or area, which has been predetermined for each segment of the mission profile. Return to the surface at this location is selected by initiating the abort command with the mode selector in the "Emergency Return" position.

As can be seen on the functional block diagram, Figure III-2-1, a third mode of abort, or trajectory modification, is available to the crew. The "Secondary Mission" is a precomputed alternate mission for use in the event that the primary mission cannot be continued because of a malfunction which does not require immediate return to the surface. This might be, for example, an off-course trajectory which is in excess of the limitations for cislunar and midcourse correction, but which does not preclude modification of the trajectory

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to an earth orbit. In addition to continuously computing the secondary mission velocity vector requirements for the guidance computer, the abort computer will present a "go, no-go" type of display to the crew to indicate the capability for changing to a secondary mission at any specific time.

The output of the computer, in addition to providing a continuous display of abort capabilities, is utilized as a command to the guidance computer when an abort command is received. The required velocity vector increments are, in turn, obtained through use of the on-board propulsion and attitude control rockets.

Upon receipt of an abort command signal, the abort computer will provide the coordinates of the predicted re-entry vehicle impact point to the telemetry system for automatic transmission to ground stations. These coordinates will be in accordance with the return mode selected.

2.1.4 Abort Propulsion System

The abort propulsion system, shown functionally on Figure III-2-1, is described in detail in Section 5.4. The solid fuel abort rockets are the prime separation devices of the abort vehicle from the booster during all phases of boost flight. The sequence of operation of the system is described in the following paragraphs. The location of the various components can be seen in Figures III-2-2 and III-2-3.

(a) Depending upon the mission phase, the abort command provides a firing signal, through the abort programmer/sequencer, to the abort separation charges. These shaped charges, described in detail in Volume VI, are located at the abort and booster adapter interfaces, as shown in Figure III-2-2 and III-2-3, for the abort vehicle and space vehicles, respectively. Firing these charges physically separates the vehicles at the specified interface.

(b) The same firing signal is utilized to fire the abort rockets. A short interval time-delay squib is inserted in the circuit to prevent these rockets from firing until complete separation is attained.

(c) During the stage S-I and S-II Saturn boost phases, the abort rocket ignition signal is utilized, through a 4.5 sec. time-delay squib, to ignite the forward and aft spacecraft structure separation charges. The delay is programmed to allow a sufficient time for the abort rockets to completely burn out. After stage S-II burnout the re-entry vehicle is not separated from the spacecraft structure immediately after abort rocket firing. A separate signal is therefore provided from the programmer to allow separation during re-entry from orbital or cislunar flight.

(d) The signal which ignites the structure separation charges is also utilized to fire the separation rockets. Again, a short interval time-delay squib is utilized to insure complete separation prior to rocket ignition.

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2.1.5 Recovery System

The recovery system, shown in functional format in Figure III-2-1, is described in Volume VI. No special sequence is planned for the abort case as the standard re-entry recovery sequence is satisfactory. The primary sensor, or initiator of the recovery sequence is, however, dependent on the phase of the mission in which operation is required. Either a timer, baroswitch, g-switch, or combination thereof are utilized as described in Section 2.1.2. Provisions for manual deployment of the recovery parachute and release after touchdown are provided as previously described in Section 2.1.1

2.1.6 Ground Station

During launch operations and the initial phase of boost flight, the ground controller will occupy a key position in the operation. The function of these ground personnel will be, for the most part, to monitor and advise the crew. However, in the event of a major malfunction, such as indication of impending booster explosion, the ground personnel will have the capability to initiate an abort by direct radio command and hard-wire links to the abort programmer/sequencer. During these phases sufficient data must be made available to enable these ground personnel to evaluate completely the performance of the complete space vehicle and boost vehicle systems. Vehicular, booster and range data will be made available through hard wire and telemetry inputs from the vehicle sensors and the AMR complex. In addition, voice communications will be continuously available between the ground controllers and the APOLLO crew.

2.2 FUNCTIONAL OPERATION — R-3 CONFIGURATION

As the R-3 abort system differs from that of the D-2 configuration only in the method of re-entry and landing maneuvers, the functional block diagram for the D-2 applies with but minor modifications.

At the crew stations backup manual controls must be provided for jettisoning the abort rockets and windshield canopy. Controls must also be provided for fin extension and flight maneuvering.

In the abort programmer/sequencer the abort rocket jettison sequence must also be changed as the R-3 configuration utilizes only 6 abort rockets, 4 of which are jettisoned at Saturn stage S-I burnout and 2 at stage S-IV burnout. No separation rockets are required.

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The recovery parachute system is utilized for emergency water landing only in the R-3 configuration. A single chute, manually deployed, is utilized, thereby alleviating the necessity for automatic sequencing of the recovery system.

2.3 ABORT PROPULSION REQUIREMENTS

The primary requirements for abort propulsion are developed as a result of the launch pad booster explosion problem. The time-distance requirements of 524 feet in 1.78 seconds are developed in Section 5.1.

2.3.1 Thrust Axis Inclination:

Thrust axis inclination is required during boost flight in order to prevent the booster, which may still be accelerating due to a malfunction in the booster shut-down system, from catching up and impacting with the abort vehicle after abort rocket burnout. For launch pad aborts relatively large inclination angles are desirable in order to obtain increased range before impact, whereas smaller angles are desirable in order to obtain higher apogee altitudes. Figure III-2-4 shows the relationship of the apogee and final chute deployment altitudes with thrust inclination, while Figure III-2-5 indicates the variation of the range to impact. As a result of these parametric studies, the selection of 20 degrees from the vertical as thrust axis inclination was made.

2.3.2 Abort Rocket Jettison

The need for high abort thrust to effect separation from the booster decreases rapidly as the burning time increases and the explosion threat is reduced. Due to the nature of explosive shock wave propagation the threat is effectively reduced to zero when the boost vehicle exceeds transonic velocity. At the same time the thrust required to overcome aerodynamic drag increases rapidly, reaching a peak at max q. Dynamic pressure then decreases fairly rapidly to approximately zero at 250,000 feet altitude. An additional requirement necessitating abort rocket thrust is the need to overcome the inherent booster acceleration at the moment of separation.

Figure III-2-6 shows, as a function of Saturn booster stage, the relative combined effects of these three acceleration requirements. As can be seen, the cumulative abort rocket requirement decreases fairly rapidly beyond the max q region so that it becomes economical to jettison the unnecessary abort rockets. In order to avoid additional complexity and

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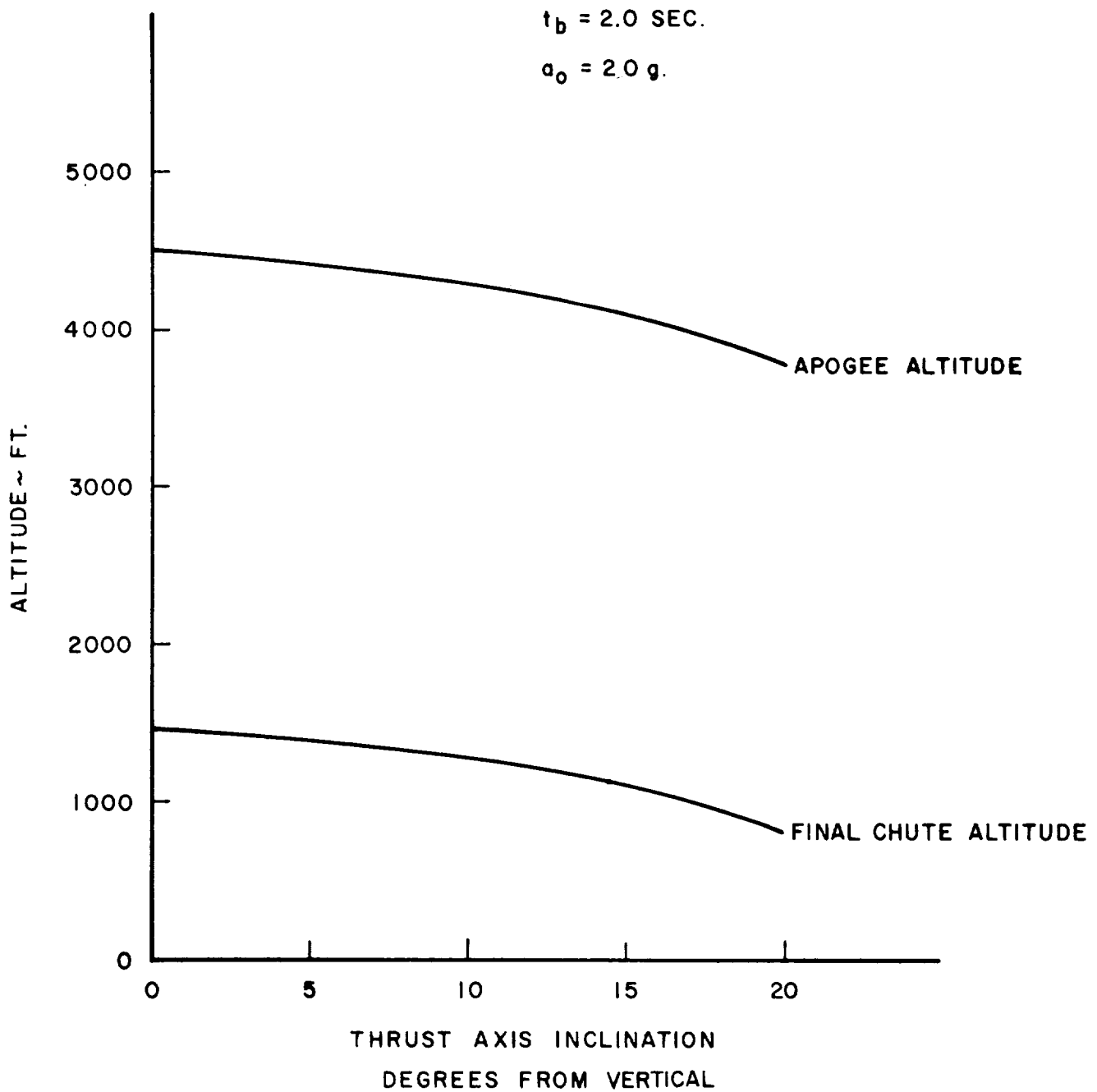


Figure III-2-4. Example of the effect of thrust axis inclination on apogee altitude and final chute deployment altitude - launch pad abort D-2 Configuration

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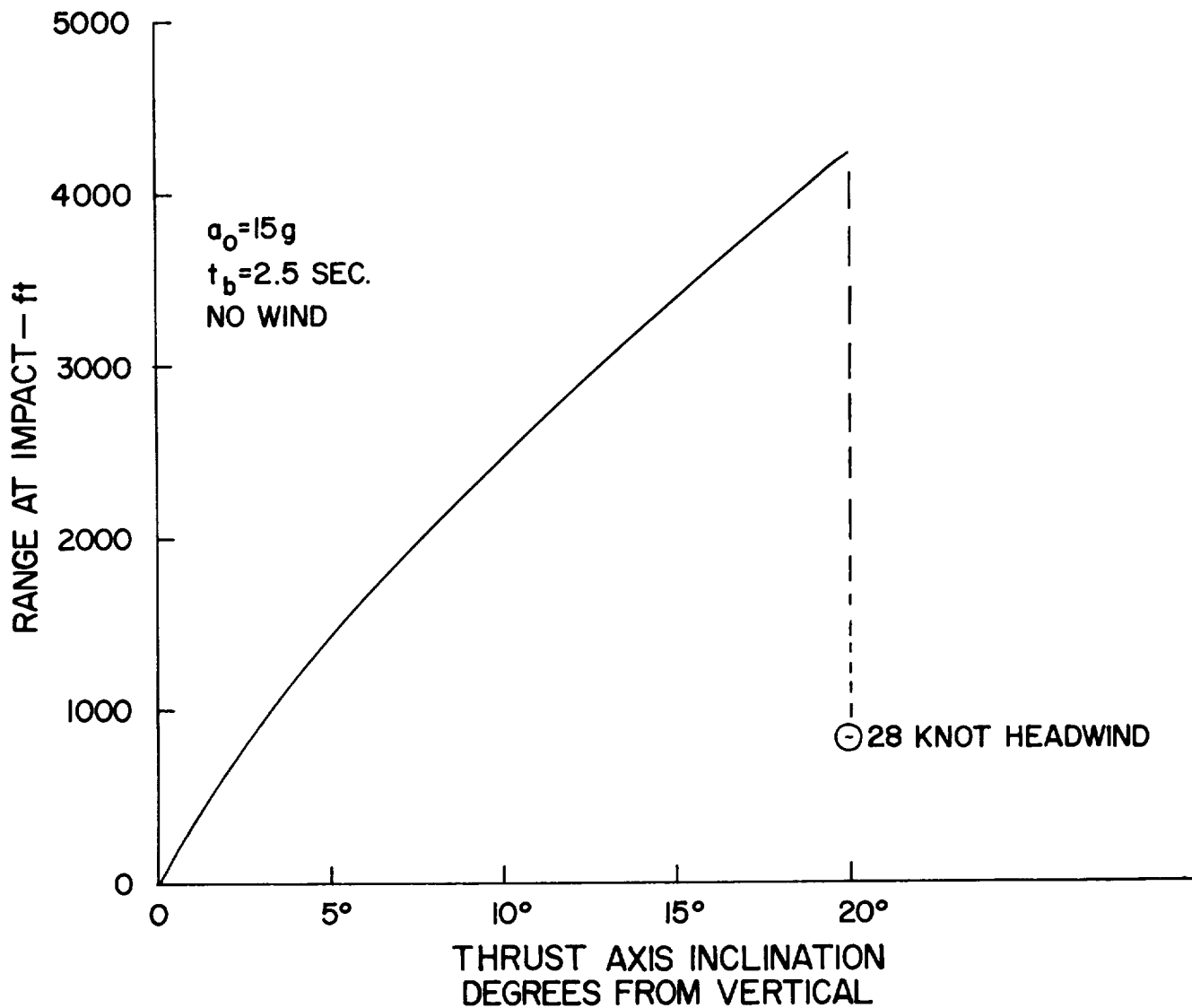


Figure III-2-5. Example of the effect of thrust axis inclination on range to impact
D-2 Configuration, launch pad abort

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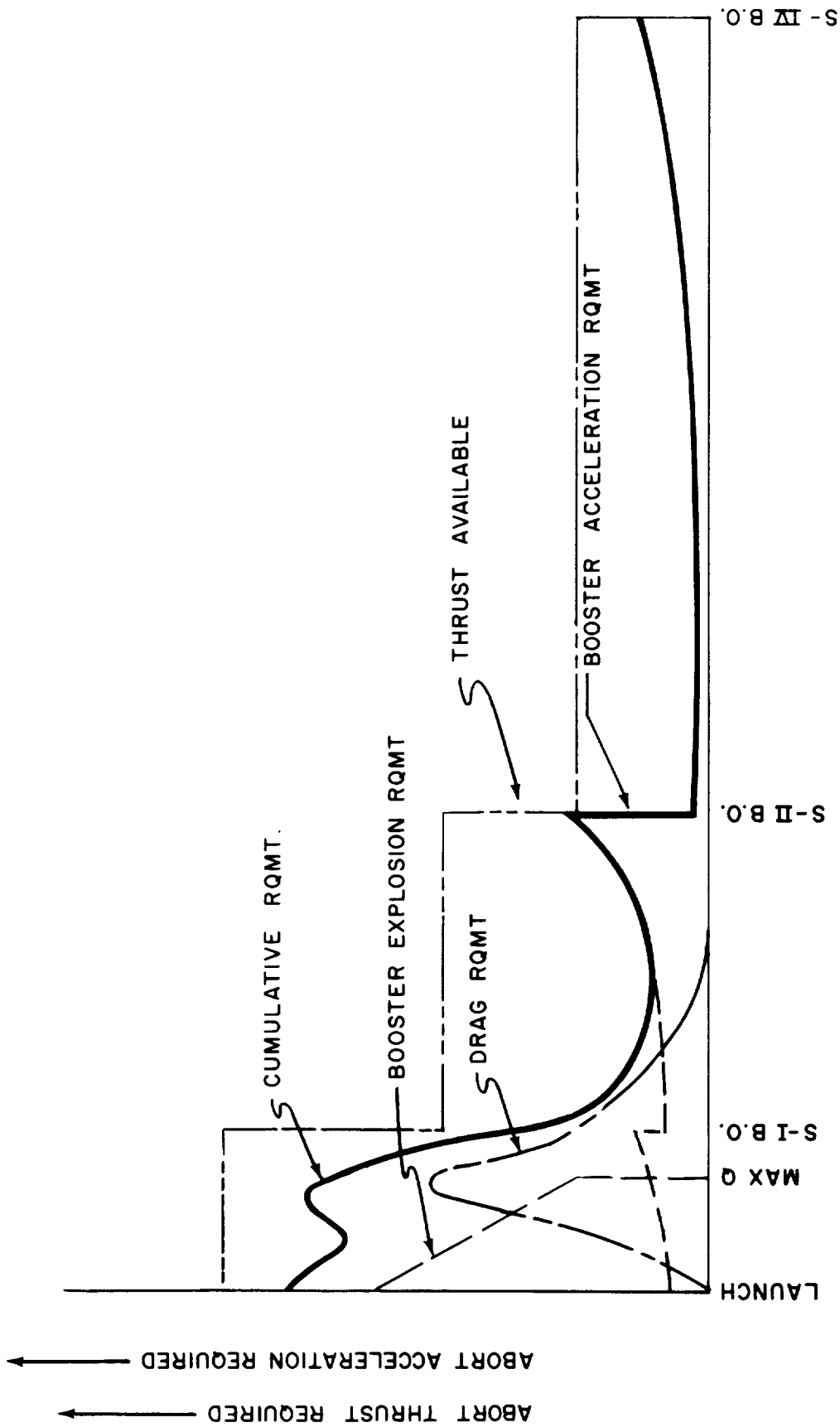


Figure III-2-6. Acceleration Requirements
D-2 Configuration

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subsequent decrease in reliability, the Saturn stage separation times, at which event commands are already available, were selected as the rocket jettison points. At S-I separation 4 of the 8 abort rockets are dropped, with a subsequent drop of 2 more at S-II burnout and separation. Figure III-2-7 indicates the available initial abort accelerations available at each of these event times.

2.3.3 Abort Rocket Orientation

Abort rocket thrust axis inclination must be provided even after launch in order that positive separation be affected in the event the booster cannot be shutdown at abort. The selected angle of 20 degrees may, however, be relaxed somewhat as the variation in range to impact with thrust inclination decreases rapidly as the vehicle velocity increases.

As shown in Figure III-2-8, the abort rocket thrust axis is physically aligned 30 degrees to the longitudinal axis of the vehicle in order to pass through the center of gravity. We must therefore orient the abort rockets circumferentially around the vehicle to compensate for this and to attain an effective thrust axis of 20 degrees.

The vertical component of the abort thrust, T_V is

$$T_V = T \cos \delta = T \cos 60^\circ \quad (1)$$

The lateral component of the abort thrust, T_L is

$$T_L = n T_V \sin \Theta \quad (2)$$

where n is the number of rockets

$$T_L = n T \cos 60^\circ \sin \Theta \quad (3)$$

The effective thrust, T_{EFF} is

$$T_{EFF} = n T \cos \beta = n T \cos 70^\circ \quad (4)$$

But $T_{EFF} = T_L$

$$n T \cos 60^\circ \sin \Theta = n T \cos 70^\circ \quad (5)$$

$$\sin \Theta = \frac{\cos 70^\circ}{\cos 60^\circ} = 0.684$$

The mean value of the abort rocket orientation required to attain a 20° thrust inclination is 43.2°

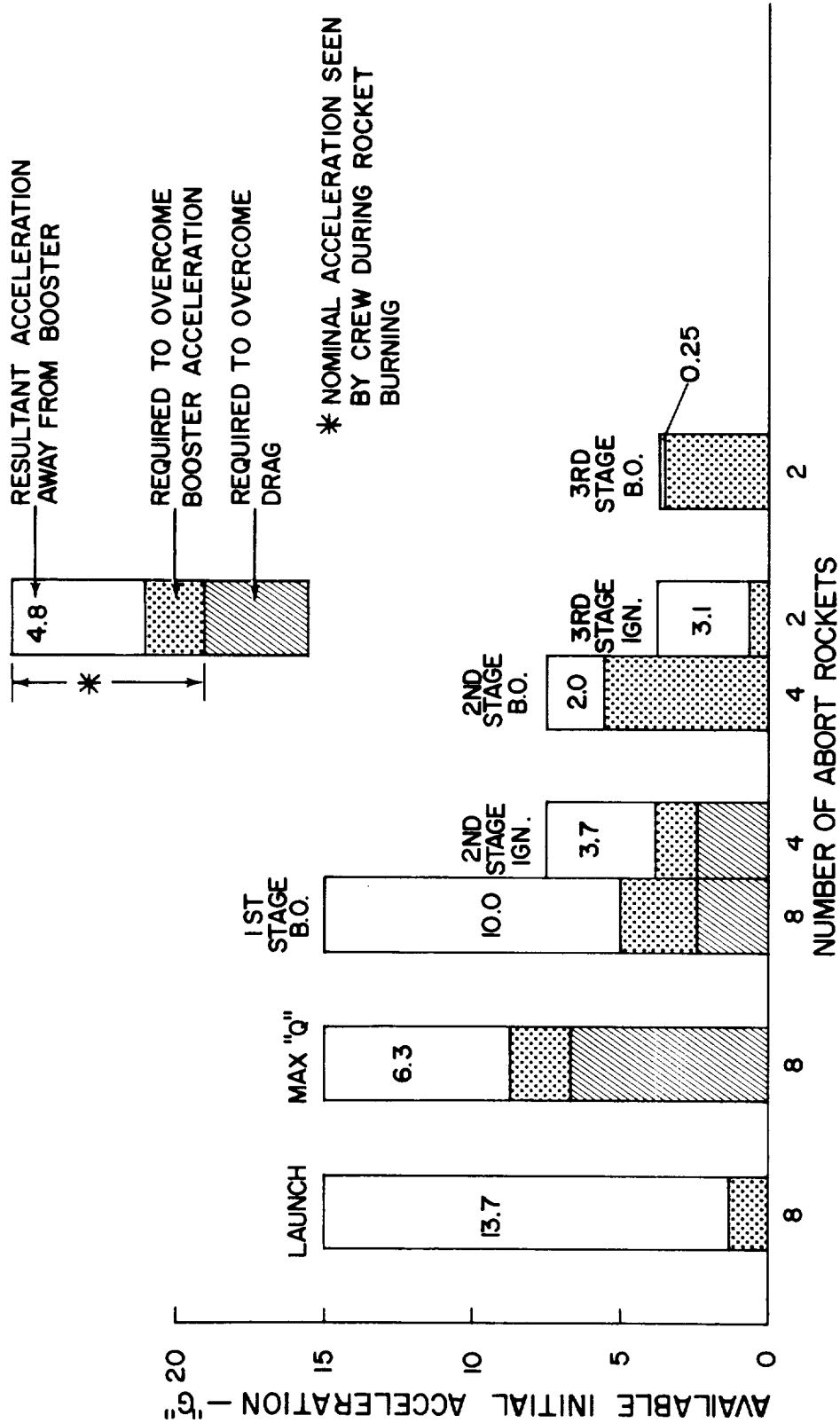


Figure III-2-7. Acceleration available during boost phases

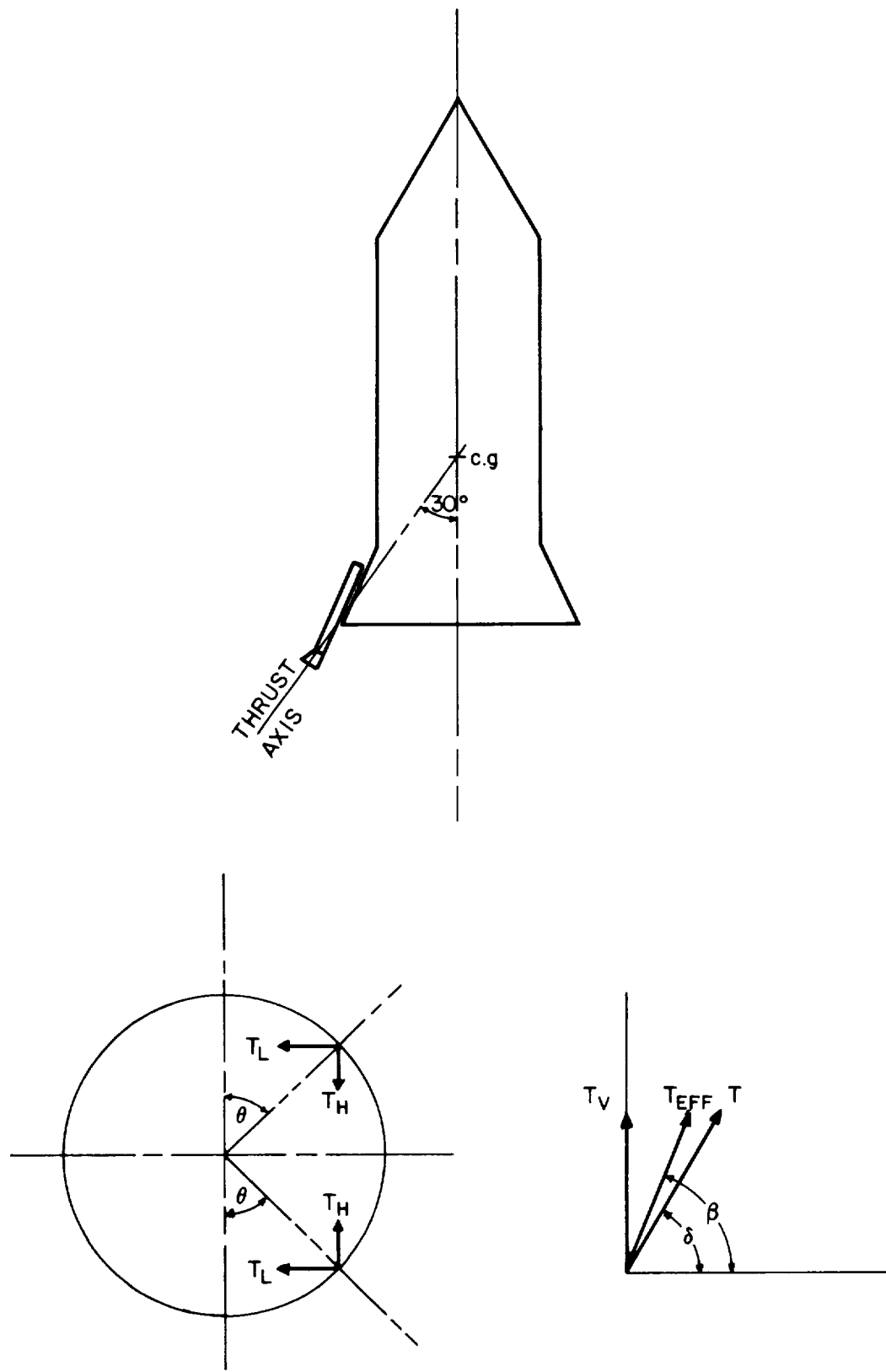


Figure III-2-8. Geometry of abort rocket thrust alignment
D-2 Configuration

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Assuming one abort rocket radius (6") spacing between the rockets, the location of each rocket can be determined from Figure III-2-9.

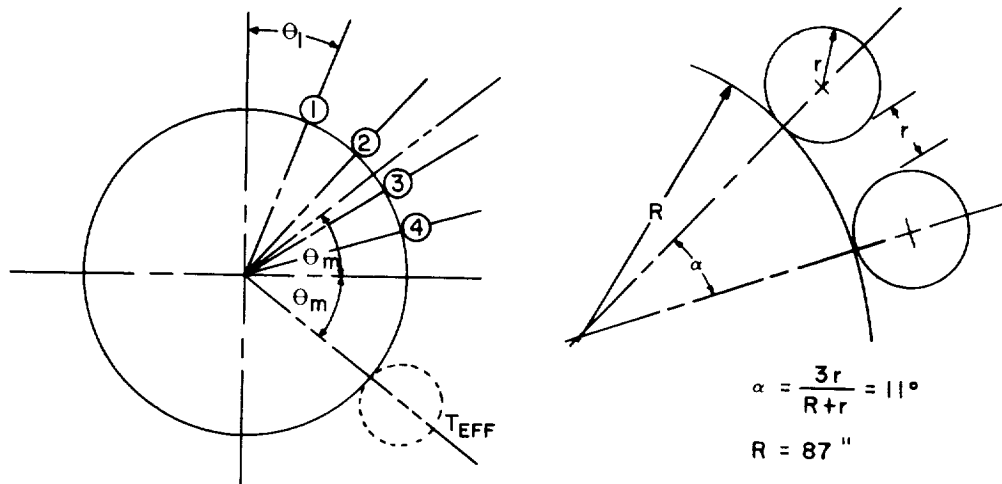


Figure III-2-9: Abort rocket orientation

$$T_L = 2T \cos 60^\circ [\sin \Theta_1 + \sin (\Theta_1 + \alpha) + \sin (\Theta_1 + 2\alpha) + \sin (\Theta_1 + 3\alpha)] \quad (6)$$

also, from eq. (5)

$$T_L = 2 n T \cos 70^\circ \quad (7)$$

therefore

$$\left[\sin \Theta_1 + \sin (\Theta_1 + \alpha) + \sin (\Theta_1 + 2\alpha) + \sin (\Theta_1 + 3\alpha) \right] \left\{ \frac{2 n T \cos 70^\circ}{2 T \cos 60^\circ} \right\} \quad (8)$$

$$\left[\sin \Theta_1 + \sin (\Theta_1 + \alpha) + \sin (\Theta_1 + 2\alpha) + \sin (\Theta_1 + 3\alpha) \right] = 0.684 n \quad (9)$$

Assigning the symbol K to the right hand side of eq. (9) a parametric plot can be made of this function with various arbitrary values of Θ_1 . From this plot, Figure III-2-10, the value of Θ_1 for

$$K = 0.684 \times 4 = 2.736 \quad (10)$$

is 27.9 degrees

At Saturn stage S-I burnout, the outer pair of each set of abort rockets (nos. 1 and 4 in figure 2-9) are dropped.

Therefore:

$$n K = \sin (\Theta_1 + \alpha) + \sin (\Theta_1 + 2\alpha) \quad (11)$$

$$2K = \sin 38.9^\circ + \sin 50.9^\circ \quad (12)$$

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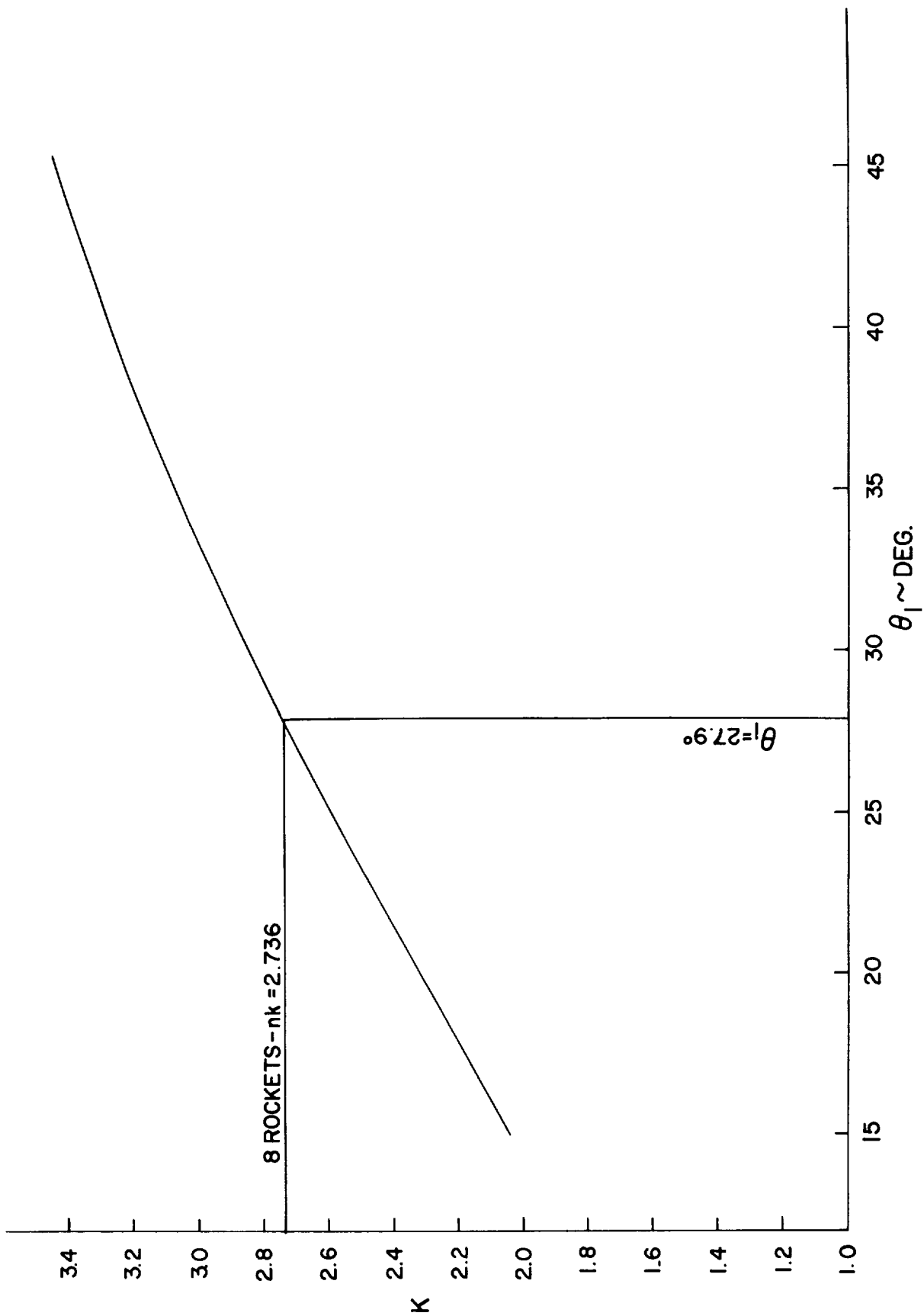


Figure III-2-10. Parametric plot to obtain abort rocket orientation

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$$K = 0.7235 \quad (13)$$

$$\cos \beta = K \cos \delta = 0.7235 \cos 60^\circ \quad (14)$$

$$\beta = 68.8^\circ, \text{ or } 21.2^\circ \text{ from the vertical}$$

At Saturn stage S-II burnout, one additional rocket is dropped from each pair (no. 3 on figure 2-9).

$$n K = \sin (\Theta_1 + \infty) \quad (15)$$

$$K = \sin 38.9^\circ \quad (16)$$

$$K = 0.628 \quad (17)$$

$$\cos \beta = K \cos \delta = 0.628 \cos 60^\circ \quad (18)$$

$$\beta = 71.7^\circ, \text{ or } 18.3^\circ \text{ from the vertical}$$

In summation, by judicious placement the abort rocket thrust axis orientation has been maintained relatively close to 20 degrees from the vertical even after 4 or 6 of the rockets have been jettisoned. Figure III-2-11 summarizes the locations selected.

2.3.4 Separation Rockets

The requirement for the large separation rockets is a result of the necessity for positive separation of the low drag forward space vehicle structure from the re-entry vehicle under high drag conditions. Once separation is affected, the low drag of this section relative to that of the re-entry vehicle assures adequate separation during ballistic flight. Conversely, the relatively high drag of the aft, or skirt section, of the space vehicle structure relative to the re-entry vehicle provides adequate trajectory spacing after physical separation is provided by small separation rockets. The large forward section separation rockets are not needed beyond the region of high q and are dropped at Saturn Stage II burnout. Small rockets, identical to the aft section separation rockets, provide separation forces after this time.

The separation distances between the forward and aft space vehicle structure sections and the re-entry vehicle, after separation of the spacecraft, were computed utilizing an average drag coefficient for the sections in tumbling flight at peak dynamic pressure. The respective separation distances of the forward section above the re-entry vehicle and the aft section below the re-entry vehicle are presented in Figure III-2-12.

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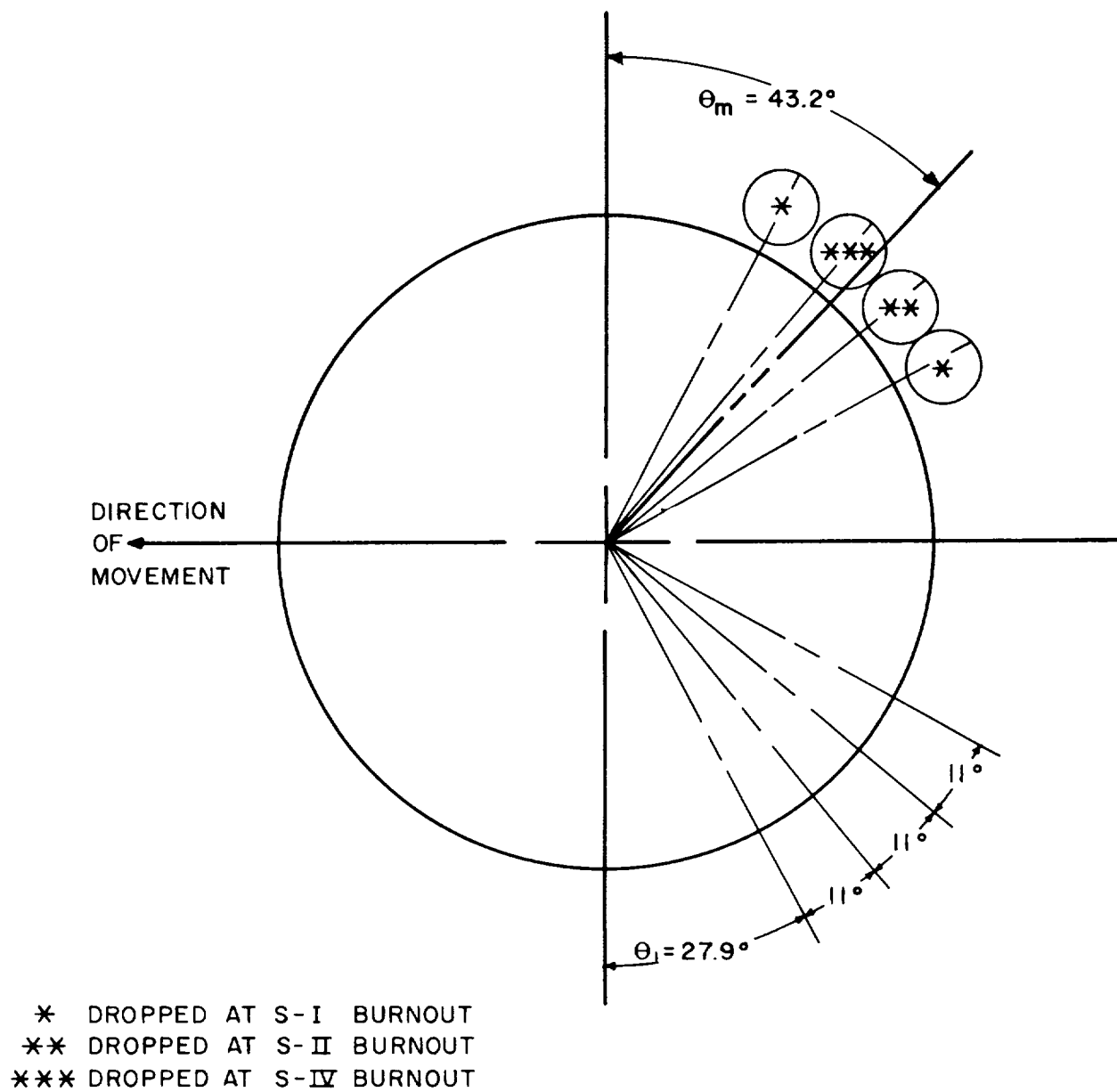


Figure III-2-11. Abort rocket orientation
D-2 Configuration

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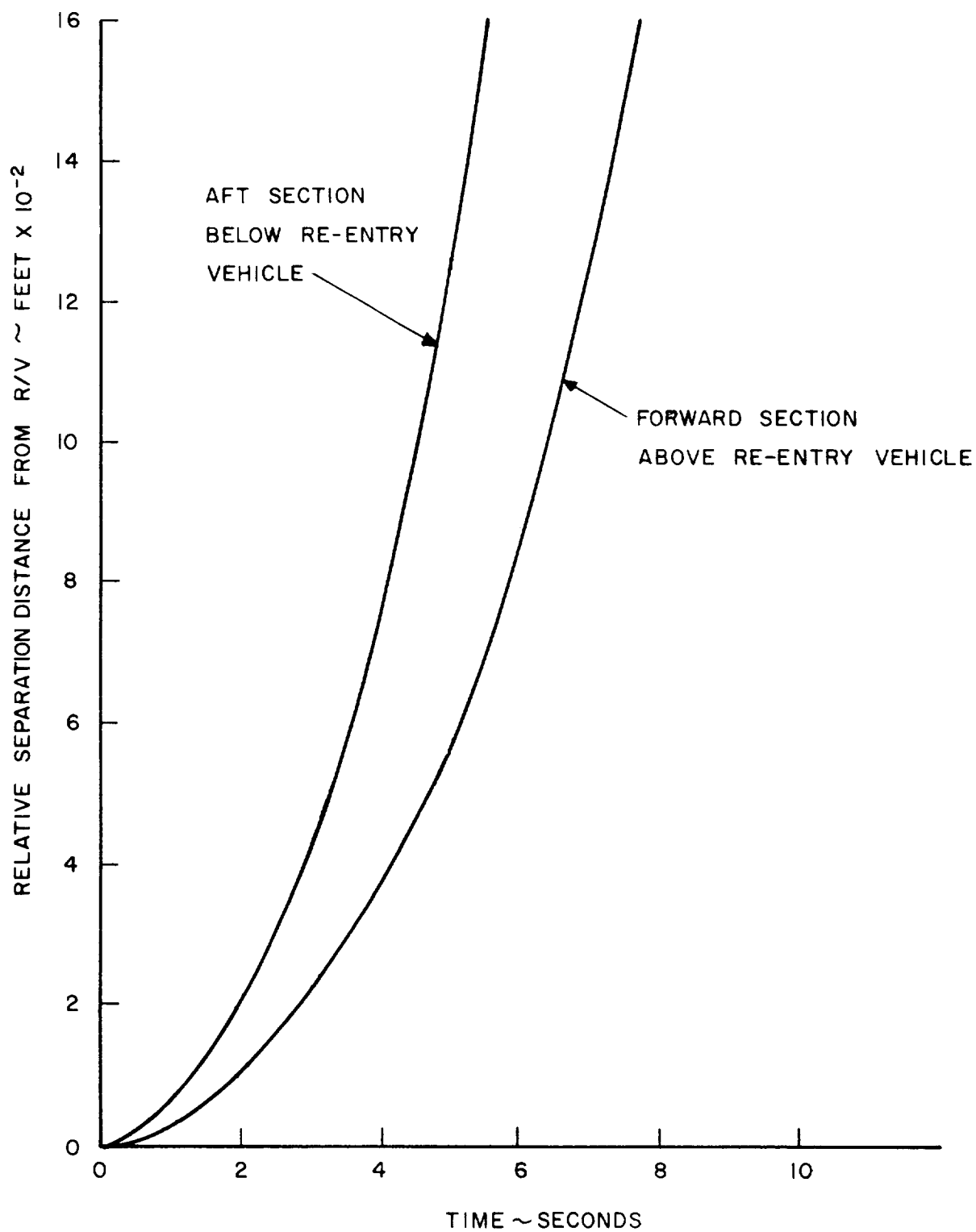


Figure III-2-12. Abort separation distances at max q
D-2 Configuration

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2.4 CREW PERFORMANCE

The role of the crew in the operation of the APOLLO D-2 vehicle and its subsystems is covered in detail in Volume V, Chapter 2.

In the event of an automatically initiated abort, his role will be that of monitor as the sequence of events will be automatically programmed. However, in the event of a component malfunction, he may, prior to operation of a secondary or redundant component, provide a manual input to the system. For example, if the baroswitch does not initiate the recovery system sequence at 25,000 feet, it may be started by the crew prior to operation by the g-switch actuated timer.

For a manual abort the crew will actuate the sequence when desired. Ground initiated aborts may be sequenced directly, thereby bypassing the crew, or passed through the crew by means of radio or telemetry link.

Beyond Saturn stage S-II burnout virtually all abort sequences will be initiated by the crew. Therefore, the mode selection is independent of the source of abort initiation and will always be selected by the crew.

2.5 ABORT ROCKET FAILURE

One of the advantages of the multiple abort rocket system is the inherent redundancy available. In the event of failure of any one of the rockets normally available for abort during any mission phase, sufficient thrust is available from the remaining rockets to assure safe abort.

Figure III-2-13 shows the abort acceleration available during the various phases of boost flight with a single abort rocket malfunction. At launch, 11.8 g initial acceleration is available which will propel the abort vehicle a distance of 649 feet in 1.78 seconds, thereby satisfying the minimum time-distance requirement of 630 feet in 1.78 seconds set by the booster explosion criteria (see Section 4.1.1). For maximum q abort, 4.4 g initial acceleration, in excess of that required to overcome drag and booster acceleration, is available. This will provide, at $t = 1.81$, a separation distance of 605 feet, which is well in excess of the minimum requirement of 381 feet (Section 4.1.2).

After max q, the only requirement for abort rocket thrust level is that it be sufficient to exceed the cumulative requirements set by aerodynamic drag and booster acceleration.

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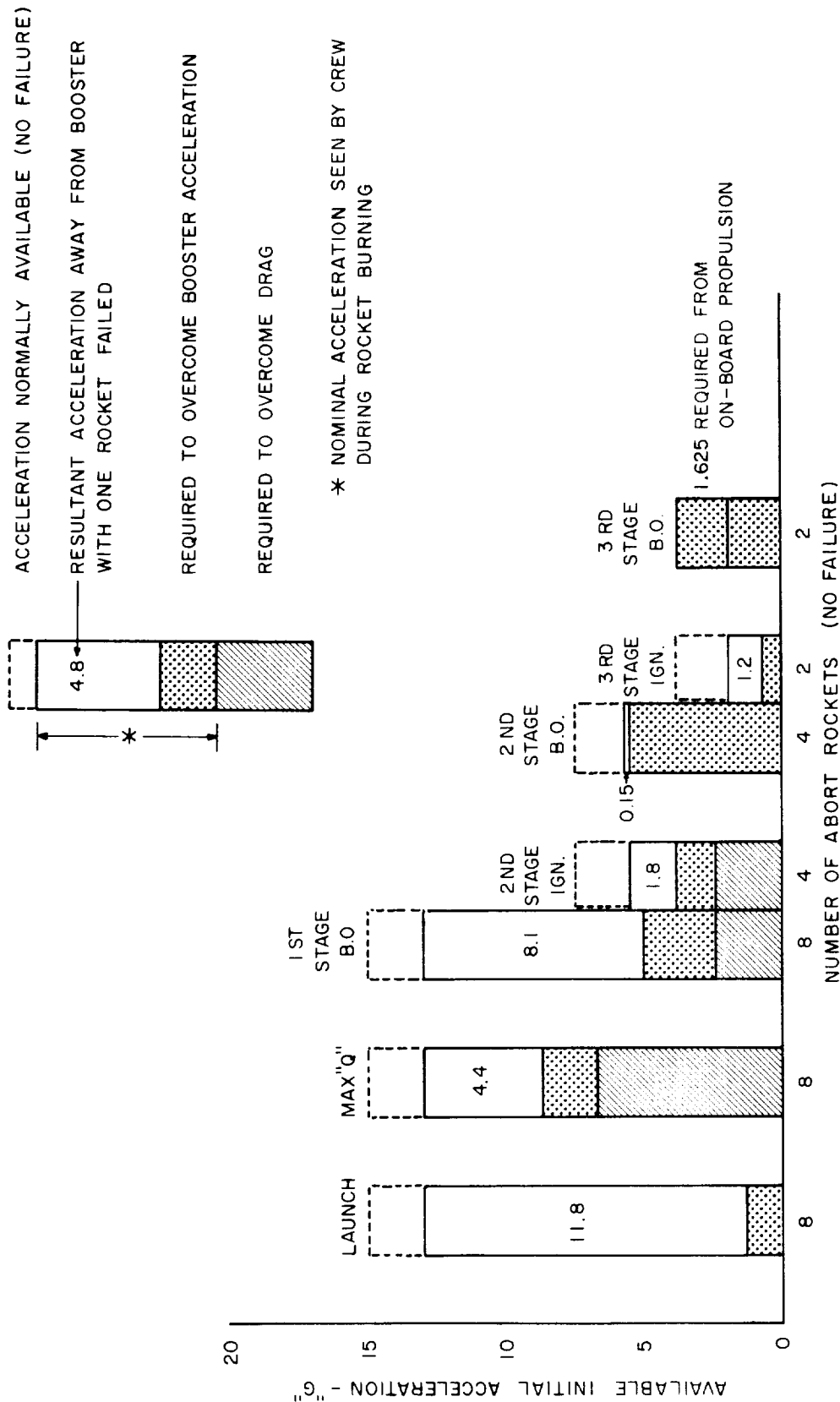


Figure III-2-13. Acceleration available during boost phases
One rocket failure

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As can be seen from Figure III-2-13, this requirement is met in all cases except at 3rd stage burnout. At this point the acceleration of the booster reaches a maximum of 3.5 g (conservatively increased from 2.55 g by taking into account the additional acceleration of the booster due to the incremental weight decrease when the APOLLO abort vehicle separates) while the single operative abort rocket provides only 1.87 g. The deficit of 1.625 g can, at this point, be easily made up by utilization of the on-board propulsion. Therefore, at no point in the boost trajectory will the failure of one abort rocket cause failure of the abort sequence.

2.6 PARACHUTE FAILURE

As detailed in Section 5.5 of this Volume and Chapter 1 of Volume VI, the normal re-entry retardation system is utilized for the D-2 abort vehicle. This system utilizes a cluster of 3 main chutes to provide final retardation to approximately 30 feet/second at impact. The effective $C_D A$ of this system is 4673 sq ft. In the event of failure of one of the 3 chutes, the effective $C_D A$ is reduced to 3115 sq ft. This provides a maximum impact velocity of 37.1 feet/second which, in the event of an abort emergency, can be tolerated by the crew.

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3.0 System Philosophy

Although the APOLLO system is designed to prevent the occurrence of critical failures we must, due to the presence of the crew, design for abort conditions which may occur during any phase of the mission. For the most part, these will occur due to propulsion and guidance-control system malfunctions; the former resulting in explosion or critical loss of thrust, and the latter in severe instability or deviation from the programmed trajectory. Although less likely, aborts may also result from stage separation malfunctions, structural failure, or from on-board emergencies arising from subsystem failures, fire, or cabin environmental control malfunctions.

3.1 MODES OF FAILURE

3.1.1 Category of Emergency

The types of emergency conditions which might be expected during an APOLLO mission are classified by the relative urgency of corrective action necessary. Studies of escape from present day launch systems caused by booster explosions have shown that the approximate minimum warning time for an impending booster explosion is in the order of 2 seconds (See Section 5.1). The time period between that at which the impending explosion is first sensed by, for example, an increase in tank pressure and the period at which the abort rocket attains full thrust is in the order of 0.5 seconds (maximum). Approximately 1.5 seconds remains, therefore, for movement of the abort vehicle away from the booster.

The categories of emergency are summarized in Table III-3-I and are further defined in the following paragraphs.

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TABLE III-3-I CATEGORIES OF FAILURE

| <u>Category</u> | <u>Immediate Action Req.</u> | <u>Result of Failure</u> | <u>Repairs or Replacement Possible</u> | <u>Redundancy Required</u> |
|-----------------|----------------------------------|---|--|--------------------------------|
| A-1 | Yes | Abort | No | No |
| A-2 | Yes | Abort, Emergency Return, or Sec- ondary Mission | No | No |
| B-1 | No | Abort or Emergency Return | No | No |
| B-2 | No | Abort, Emergency Return or Secondary Mission | No | No |
| C-1 | No | Emergency Return or Secondary Mission | Yes | Yes |
| C-2 | No | No Change | Yes | No |
| D | No | No Change | Yes | Yes |

NOTE: Emergency systems will be operative in the event a redundancy is not required.

Category A-1

Emergency conditions which require immediate action and which will result in a complete abort of the mission; e.g., explosion of the booster during launch.

Category A-2

Those emergency conditions which require immediate action which will result in a deviation from the planned mission objectives to a secondary objective, e.g., insufficient final stage boost which will allow deviation from a lunar mission to a secondary earth orbit mission.

Category B-1

Major emergencies and systems failures which will result in the complete abort of the mission. These differ from Category A-1 failures in that sufficient time is available to allow for an analysis of the emergency, by either crew members or ground monitors, and manual initiation of the abort procedure; e.g., power supply failure. The availability of redundant systems will not effect this category of emergency.

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Category B-2

Major emergencies and systems failures which allow sufficient time for analysis of the problem and manual initiation of a procedure which will result in a change to a secondary mission objective; e.g., insufficient fuel remaining to make necessary trajectory correction and still complete lunar mission. The availability of redundant systems will not affect this category of emergency.

Category C-1

Equipment failure in a major system but where a redundant system is available. If the failed component is not repairable or replaceable in flight, the Category C-1 failure will result in an emergency return or change to a secondary mission objective in order to shorten the mission time; e.g., loss of pressure in one tank of a multi-tank oxygen supply system.

Category C-2

Equipment failure, which although compromising the mission objectives somewhat, does not require a deviation from the planned profile. If a Category C-2 failure occurs in a major operating system repair of the malfunctioning equipment must be possible; e.g., voice communications failure. If a Category C-2 failure occurs in a nonoperating system and is not repairable, the mission may still be completed with some degradation in expected results; e.g., partial failure of mission instrumentation equipment.

Category D

Equipment or system failure external to the APOLLO flight vehicle. This may include range instrumentation, GSE, etc. During the pre-launch phase this will result in a count-down hold. At other times the mission will not be affected as it is expected that redundant ground based systems and sufficient spare parts will be available.

3.1.2 Abort Command

Whenever feasible the APOLLO vehicle will be under control of its crew. This is specifically required in order to optimize the mission observation function. Command during abort, while attempting to comply with this concept, will necessarily be delegated to the command function at which the maximum decision making capability is available at that time.

The location of the best data available for necessary decisions varies according to the nature of the decision and the phase of the mission. For example, overriding control for abort during pre-launch or launch must, for critical emergencies, reside in the ground installation. While the crew will always have the capability of initiating aborts on their own initiative, no veto power may be allowed to be exercised by them due to lack of knowledge of the total situation coupled with the high-g environment of the launch phase. For other emergency situations control may be in the hands of the crew, in the hands of automatic vehicle-borne equipment, or in the hands of ground equipment or personnel.

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3.1.3 Types of Failure

The following listing delineates the system or subsystem failures which have been considered in this study.

- Booster explosion
- Loss of thrust
- Insufficient thrust
- Booster fails to ignite
- Stage separation failure
- Power supply failure
- Guidance system failure
- Control failure
- Major structural failure
- Fire in re-entry vehicle
- Fire in mission module
- Leaks
- Meteoroid penetration
- Telemetry failure
- Communication failure
- Tracking beacon failure
- Vehicle instrumentation/display failure
- Cabin gaseous control failure
- Cabin thermal control failure
- Bio-instrumentation failure
- Mission instrumentation failure
- Human failure-physiological
- Human failure-psychological
- On-board propulsion failure
- Failure external to vehicle

3.2 ABORT CRITERIA

The detection of an emergency condition requires action within a finite time period by the command function. The various emergency conditions listed require different action according to the mission phase in which they occur. In the following paragraphs the various failures, malfunctions, or emergencies listed in Section 3.1.3 are classified according to the emergency categories developed in Section 3.1.1. These criteria are also tabulated, as a function of mission phase, in Tables III-3-II to III-3-V.

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3.2.1 Pre-Launch

Virtually all failures in this phase may be classified as Category B-1 situations where the failure is detected by either the ground monitor or crewmember. He then manually initiates the abort procedure which, in this case, may be a countdown hold or an evacuation of the vehicle. An exception to this would be the case of a booster explosion which would require immediate abort action. Table III-3-II summarizes the failure conditions considered in the pre-launch phase together with the resultant actions.

3.2.2 Saturn Boost Phases

The categories of failure which may occur during the boost phases are summarized in Table III-3-III. There is essentially no difference in the action to be taken for a particular emergency relative to the stage of boost in which it occurs. The three exceptions to this rule are meteoroid penetrations, insufficient thrust, and failure of the booster to ignite. Meteoroid penetrations are classified as Category A-2 failures during the second and third stages of powered flight but are not considered applicable during the initial phase due to the low altitude. Loss of thrust, or insufficient thrust, is considered a Category A-1 failure if it occurs during first or second stage boost but is classified Category A-2 during third stage boost due to the possibility of changing from a lunar to earth orbital mission at that time. Failure of the second or third stage boosters to ignite constitutes a Category A-1 emergency whereas failure of the first stage to ignite is classified Category B-1.

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TABLE III-3-II PRE-LAUNCH PHASE EMERGENCIES

| <u>Emergency</u> | <u>Category</u> | <u>Abort Initiation</u> | <u>Method of Abort</u> |
|---|-----------------|-----------------------------|----------------------------|
| Booster Explosion | A-1 | Automatic | Eject using escape rockets |
| Communications Failure | B-1 | Manual-Vehicle/Ground | Hold or Scrub |
| Telemetry Failure | B-1 | Manual-Ground | Hold or Scrub |
| Tracking Beacon Failure | B-1 | Manual-Ground | Hold or Scrub |
| Vehicle Instrumentation/Display Failure | B-1 | Manual-Vehicle | Hold or Scrub |
| Mission Instrumentation Failure | B-1 | Manual-Ground | Hold or Scrub |
| Guidance System Failure | B-1 | Manual-Ground | Scrub |
| Control Failure | B-1 | Manual-Ground | Scrub |
| Cabin Gaseous Control Failure | B-1 | Manual-Vehicle | Scrub |
| Cabin Thermal Control Failure | B-1 | Manual-Vehicle | Scrub |
| Power Supply Failure | B-1 | Manual-Ground | Hold-Scrub |
| Bio-Instrumentation Failure | B-1 | Manual-Vehicle/Ground | Hold-Scrub |
| Mission Module Fire | B-1 | Manual-Vehicle | Scrub |
| Major Structural Failure | B-1 | Manual-Vehicle/Ground | Scrub |
| Leaks | B-1 | Manual-Vehicle | Scrub |
| Fire in Re-entry Vehicle | B-1 | Manual-Vehicle | Scrub |
| Human Failure-Physiological | B-1 | Manual-Vehicle | Scrub |
| Human Failure-Psychological | B-1 | Manual-Vehicle | Scrub |
| Failure External to Vehicle | D | Manual-Ground | Hold or Scrub |

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TABLE III-3-III BOOST PHASE EMERGENCIES

| <u>Emergency</u> | <u>Category</u> | <u>Abort/Emergency Return Initiation</u> | <u>Method of Abort/Emergency Return</u> |
|---|-----------------|--|--|
| Booster Explosion | A-1 | Automatic | Eject using escape rockets, automatically activate recovery aids and emergency systems. If beyond S-II stage burnout utilize on-board propulsion to change trajectory. |
| Power Supply Failure | A-1 | | |
| Loss of Thrust/Insufficient Thrust | A-1* | | |
| Booster Fails to Ignite | A-1** | | |
| Stage Separation Failure | A-1 | | |
| Guidance System Failure | A-1 | | |
| Control Failure | A-1 | | |
| Major Structural Failure | A-1 | | |
| Fire in Mission Module | A-2 | | |
| Fire in Re-Entry Vehicle | A-1 | | |
| Telemetry Failure | A-2 | Automatic (1) Automatic (1) | Automatic reprogramming of guidance system to initiate earth orbit mission. Manual control after arriving in orbit. |
| Leaks | A-2 | | |
| Meteoroid Penetration | A-2*** | | |
| Human Failure-Physiological | B-1 | Manual Manual | Initiate preprogrammed change in guidance and booster control system to effect rapid return to earth. Change to earth orbit mission. Perform on- board maintenance. If not successful in repairing, initiate action to return to earth. |
| Human Failure-Psychological | B-1 | | |
| Communications Failure | C-1 | Manual Manual Manual Manual Manual | Initiate preprogrammed change in guidance and booster control system to effect rapid return to earth. Change to earth orbit mission. Perform on- board maintenance. If not successful in repairing, initiate action to return to earth. |
| Tracking Beacon Failure | C-1 | | |
| Vehicle Instrumentation/Display Failure | C-1 | | |
| Cabin Gaseous Control Failure | C-1 | | |
| Cabin Thermal Control Failure | C-1 | | |
| Bio-instrumentation Failure | C-2 | None | None. On-board maintenance |
| Failure External to Vehicle | D | None | Ground Maintenance |

* Category A-2 during 3rd stage boost

** Category B-1 during 1st stage boost

*** Not applicable during 1st stage boost

(1) Crewmember action to repair/reduce
leaks taken after earth orbit is
established.

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3.2.3 Cislunar Flight (Also applicable to earth orbital mission phases)

The categories of failure which may occur during cislunar flight are summarized in Table III-3-IV. Most failures occurring in this phase will result in an emergency return or modification of the mission to an earth orbit with re-entry time dictated by the nature of the emergency. Although a number of Category B-1 emergencies are listed in the table, the procedure utilized will be to modify the trajectory to earth if possible, with subsequent re-entry while operating on an emergency system.

3.2.4 Re-entry, Landing, and Recovery

Emergencies in these mission phases are summarized in Table III-3-V. All failures in these phases are classified Category B-2, however, no major change in the mission profile will be possible. In the event of an emergency, operation will continue on emergency or redundant systems. On-board maintenance cannot be performed during these phases because of the necessity for the crew to be in protective restraint.

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TABLE III-3-IV CISLUNAR FLIGHT EMERGENCIES

| <u>Emergency</u> | <u>Category</u> | <u>Abort/Escape Initiation</u> | <u>Method of Abort/Escape</u> |
|---|-----------------|--------------------------------|---|
| Stage Separation Failure | B-1 | Manual | Initiate preprogrammed change in guidance and propulsion to obtain earth orbit trajectory at proper angle for re-entry. Use emergency systems during this period. |
| Guidance System Failure | B-1 | Manual | |
| Control Failure | B-1 | Manual | |
| On-Board Propulsion Failure | B-1 | Manual | |
| Power Supply Failure | B-1 | Manual | |
| Major Structural Failure | B-1 | Manual | |
| Fire in Re-entry Vehicle | B-1 | Manual | |
| Human Failure-Physiological | B-1 | Manual | |
| Human Failure-Psychological | B-1 | Manual | |
| Mission Module Fire | B-1 | Manual | |
| Leaks | B-1 | Manual | |
| Meteoroid Penetration | B-2 | Manual | |
| Communications Failure | B-2 | Manual | |
| Telemetry Failure | C-1 | Manual | |
| Tracking Beacon Failure | C-1 | Manual | |
| Vehicle Instrumentation/Display Failure | C-1* | Manual | Change to earth orbit mission. Perform on-board maintenance. |
| Cabin Gaseous Control Failure | C-1 | Manual | |
| Cabin Thermal Control Failure | C-1 | Manual | |
| Mission Instrumentation Failure | C-2 | None | None. Perform on-board maintenance. |
| Bio-Instrumentation Failure | C-2 | None | |
| Failure External to Vehicle | D | None | |

*C-2 failure in earth orbit mission

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TABLE III-3-V RE-ENTRY, LANDING, AND RECOVERY EMERGENCIES

| <u>Emergency</u> | <u>Category</u> | <u>Abort/Escape Initiation</u> | <u>Method of Abort/ Escape</u> |
|---|-----------------|---|---|
| Communication Failure | B-2 | Initiate Emergency Operation of Affected System | No trajectory change. Operate on emergency systems. |
| Telemetry Failure | B-2 | | |
| Tracking Beacon Failure | B-2 | | |
| Vehicle Instrumentation/Display Failure | B-2 | | |
| Control Failure | B-2* | | |
| Cabin Gaseous Control Failure | B-2* | | |
| Cabin Thermal Control Failure | B-2* | | |
| Module Separation Failure | B-2* | | |
| Bio-instrumentation Failure | B-2 | | |
| Structural Failure | B-2** | | |
| Leaks | B-2* | | |
| Recovery Aids | B-2*** | | |

*During re-entry phase only.

**During re-entry and landing phases only.

***During landing and recovery phases only.

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3.3 EMERGENCY DETECTION

Detection of emergency conditions and malfunctions will be accomplished by sensing devices which are capable of measuring the abnormal conditions that produce the requirement for abort. These sensors, together with necessary circuitry, must be simple, reliable, and cover the maximum number of possible malfunctions. In addition, wherever possible, the abort sensors and circuits should be entirely independent of the normal flight and vehicle instrumentation. In most cases, the normal flight instruments can be employed to verify the abort circuit signal and indicate specific abnormal conditions which are not covered by the abort sensors. A separate power source should be provided for the emergency detection circuit. Some of the specific conditions to be sensed and the problems associated with them are discussed in the following paragraphs.

3.3.1 Propellant Tank Pressure Sensors

Propellant (fuel and liquid oxygen) tank pressures may be detected and reported by conventional and reliable pressure sensors. The sensors must detect critical underpressures, pressure differentials between LOX and fuel tanks, and overpressures. The exact methods of incorporating the various sensors on their relative components will be determined in a subsequent part of the program.

3.3.2 Loss of Thrust

Thrust loss can be monitored by a pressure sensor located in the walls of the expansion section of each combustion chamber. The sensors cannot be placed at the throat or in the gas stream because of the high temperatures existing at these points. Separate sensors in the piping ahead of the combustion chambers are unnecessary, since any serious malfunction resulting in a loss of thrust would be reported by the sensor at the expansion section.

3.3.3 Vehicle Instability

A completely separate sensor source for each potential malfunction provides the maximum in reliability for the emergency detection system. However, if this concept were applied to detect instability, the sensor system would approach the complexity, size, and weight of the normal guidance system stable platform. Therefore, to reduce weight, stability information should be obtained from the roll, pitch, and yaw rate gyros

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of the vehicle inertial guidance system. When predetermined values which would lead to severe instability are exceeded, the abort sequence is initiated automatically.

3.3.4 Trajectory Deviation

It would be impractical to provide a separate sensor in the emergency detection and warning system to detect a trajectory deviation. In the event that the vehicle should deviate from the programmed trajectory, the crew would be informed of position, altitude, or velocity errors by the vehicle navigation display panel. In addition, the ground stations which track the vehicle and determine its position would report the condition to the crew. However, the possibility of a communications failure demands that the airborne equipment be capable of providing navigational data.

Values of the velocity, altitude, and flight path angle up to the point of injection, or at any succeeding point on the trajectory, may be compared by the crew with those required for mission completion, in order to evaluate the existing situation. In the event that the navigation function of the airborne guidance system and the ground tracking system both failed, conventional navigation techniques may be employed. Simple slide rules relating vehicle performance capability to present velocity and altitude would serve as a back-up. A duplicate airborne guidance system would be desirable, but the weight penalty would be excessive.

3.3.5 Stage Separation

The failure of a stage to separate will be detected by mechanically-actuated interlock relays which are a part of the warning circuit. These relays, normally open, would close only when stage separation was complete. They would be activated by means of programmer control circuits, just prior to staging. The sensor relays should be located at the stage connection points and at electrical and other service connections.

3.3.6 Structural Failure

The detection of structural failures of the total vehicle by means of instrumentation would be impractical. Although it may be possible to strategically locate sensors at points of expected stress concentration, it would be impossible to instrument the structures for all possible failures. Therefore, sensors will not be provided in the emergency detection and warning system to detect structural failure.

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3.3.7 Electrical Power Failure

The loss of electrical power output will be detected by means of a normally open relay with vehicle power utilized to maintain the relay contacts in the closed position. When the voltage drops below what is considered a minimum for reliable electrical systems operation, the sensor relay contacts open, causing a warning signal to be displayed in the vehicle.

3.3.8 Fire Detection

Fires will be detected by strategically-located area surveillance-type sensors. The areas most sensitive to the fire hazard are those in the vicinity of the fuel storage tanks, cabin conditioning, and escape-associated equipment. Overheat detection devices installed on the power supply and other high-temperature machinery will warn of excessive temperatures that may weaken structures or ignite combustible materials.

3.3.9 Cabin Atmosphere

a. Decompression

Decompression may be sensed by means of a simple bourdon-tube pressure sensor located in the vehicle. However, the decision to abort depends not only upon the actual pressure but also upon the rate of decompression and the ability of the cabin atmosphere conditioning system to make up the air leakage for the duration of the mission. Therefore, the cabin pressure sensor must be integrated into a flow meter system which will inform the crew of the time remaining before a habitable atmosphere can no longer be maintained.

b. Pollution

Gas analyzers located in the capsule will continually determine the amounts of CO₂, CO, HF, and Cl₂ present and indicate when unsafe concentrations exist. However, any instrument for detecting CO₂ concentration is probably less reliable than the CO₂ absorption portion of the air conditioning system. Therefore, a considerable reliance must be placed upon the crew's ability to recognize preliminary symptoms of the effects of atmospheric contamination.

3.3.10 Abort Actuation Controls

In previously discussed section on abort command, it was shown that the abort circuit may be energized by either manually closing a switch or a remote signal (from the

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ground or from automatic airborne equipment) which closes an escape initiation relay. An escape initiated by a crew member requires some intentional, physical act by him, such as pulling a conventional hand-grip lever. The movement of a protected switch on the control console or the actuation of a small button or lever on a control stick are other methods of manually initiating the escape.

The crew must be capable of initiating the abort even under the most severe vibration and acceleration conditions where the slightest physical movement requires great effort. Results of related environmental tests must be studied to determine the optimum method and location of the abort control considering the relationship of this control to the other vehicle controls.

3.4 WARNING SIGNALS

The crew and ground personnel must be alerted when an emergency which requires an abort decision occurs. The signal system by which the alert is accomplished may be arranged to present varying degrees of information. A highly sophisticated warning system would automatically indicate in considerable detail the emergency area and the degree of emergency. Such a system would be heavy and the high degree of complexity would reduce its reliability.

A simpler and more reliable arrangement would employ a single warning light for both the ground-based and airborne display panels. It would be advisable to employ an audible warning device, such as a bell or horn, in conjunction with the light. For Category A emergencies, the warning device simply indicates that an automatic abort is being initiated. When the warning device signals a Category B, C or D emergency, the crewmen or ground monitors must scrutinize their respective instrument panels in order to determine the type of malfunction in order to evaluate the severity of the emergency. With such a warning system, the warning signal would remain on as long as the sensor was exposed to an abnormal value. If the emergency was of the type where a delayed abort, or emergency return, was warranted, a warning signal release, which would deactivate the audible signal, would be desirable to permit subsequent emergencies to be properly reported by the system. If complete communications are maintained between ground and vehicle, most Category B and C emergencies will be evaluated jointly by the ground-based monitor and the crewmen. This voice communication link may be construed to be a part of the warning system.

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In the event that voice communications failed but telemetry signals were reliably transmitted, an abort might be initiated from the ground without prior knowledge of the crew. Such unilateral action, although possible, is highly improbable because doubt might exist as to the true conditions aloft. It is more likely that the crew will initiate the abort without consultation from the ground station. The first indication to the ground station that an abort had been made would be the telemetric report that the abort vehicle had separated. This signal is not a part of the detection and warning circuit; however, it has a function in the abort sequence in that it serves as a warning signal to the ground station to alert the tracking stations and rescue crews, and it aids in determining the emergency landing point by informing the ground stations of the exact time of abort and predicted impact location as determined from the abort computer.

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4.0 Trajectories

4.1 SATURN C-2 BOOST PHASE

4.1.1 Launch Pad Abort

Launch pad abort trajectories were parametrically studied in order to determine the optimum thrust/burning time combination for the solid propellant abort rockets.

4.1.1.1 CRITERIA

The basic parameter affecting launch pad abort is the necessity for moving the manned vehicle a sufficient distance from an exploding booster in order that the resulting shock wave does not exceed structural limitations. However, additional criteria and limitations must necessarily be considered, weighted, and traded-off in order to optimize the system. For this study these criteria included the recovery system, obstacle avoidance, acceleration limitations, weight penalties, and compatibility during subsequent mission phase operations.

- a. Booster Explosion-The specific requirements placed on the abort system in the event of a booster explosion are that the vehicle be displaced a radial distance of 524 feet from the top of the Saturn C-2 vehicle within a maximum of 1.78 seconds. (A detailed discussion of the booster explosion problem is presented in Section 5.1.) As the APOLLO/Saturn S-IV interface is approximately 175 feet above the surface (Ref 1) the slant range of the APOLLO vehicle must be a minimum of 699 feet above the surface at this specified time.
- b. Recovery System Operation - Proper operation of the D-2 configuration recovery system requires initiation of the parachute deployment sequence at a sufficient height to insure that final stage chute deployment be completed at a sufficient altitude to assure safety of the crewmembers. For this study the apogee altitude of the launch pad abort trajectory was selected as the point for initiating the parachute deployment sequence. At this point, in addition to having a maximum of altitude, the dynamic pressure is at a minimum, but still of sufficient magnitude to insure proper parachute deployment. Emerging from the abort vehicle two seconds after rocket burnout, the re-entry vehicle, with heat shield oriented to the rear, has had sufficient time to become aerodynamically re-oriented and stabilized in the proper position for parachute deployment. The system selected and

Ref 1 Anon., Preliminary Saturn C-2 Information. NASA Space Task Group; Langley Field, Va., Feb 20, 1961.

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described in Section 4.1.1.3, utilizes a separation actuated timer which initiates the chute deployment system 12.5 seconds after abort rocket firing.

In order to improve reliability it was decided to utilize the normal re-entry parachute deployment sequence for the abort condition. The only change required is the substitution of the previously mentioned timer in lieu of the baroswitch to initiate deployment of the parachute. The recovery system operating sequence is detailed in Section 5.5.

The minimum altitude selected for final stage parachute deployment was 1000 feet. This provides a sufficient margin of safety for contingencies in the parachute deployment sequence. In addition, some degree of local obstacle avoidance capability is provided at this altitude utilizing the parachute system described in Section 5.5.3. A maximum altitude for final chute deployment was also considered, but to a lesser degree. Too great an altitude will allow the slowly descending vehicle to drift back to, or beyond, the launch area with an onshore wind blowing. The selected maximum altitude was 1500 feet.

Proper recovery system operation imposes few restrictive parameters on the R-3 vehicle abort system. The vehicle must attain sufficient velocity to provide the necessary glide capability. This is attained, and considerably exceeded, in meeting the booster explosion time-distance requirements.

c. Obstacle Avoidance - For the D-2 vehicle, the local obstacle avoidance problem is best solved by providing a sufficient down range increment, with the abort rockets, to insure landing in the ocean near the launch area. This increment is provided by offsetting the abort rocket thrust to attain an inclined thrust axis through the c.g. of the vehicle (see Section 2). In the parametric studies considered, a trade-off of the increased range obtained by increasing this angle (from the vertical), was made against the resulting decrease in apogee altitude.

An additional factor considered was the possibility of drifting back towards the launch pad as a result of an on-shore wind. Studies of the effect of wind on the selected system are shown in Section 4.1.1.2. The wind velocity used was 28 knots (47.25 fps), which is the maximum allowable for Saturn launch (Ref 2).

d. Acceleration Profile - The performance requirements of the various abort system elements are demanding for any abort requirement that may occur during all phases of the mission profile. The conditions which impose the most severe requirements, and which establish the design criteria for the acceleration means of the abort system, exist

Ref 2 Personal communication from Mr. James W. Carter, Future Projects Office, Marshall Space Flight Center, April 13, 1961.

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during the launch and initial boost phases when the vehicle altitude and velocity are at, or near, zero.

If, during launch and ascent, an explosion in any of the vehicle boost units is impending, it is imperative that the abort vehicle be removed from the area of the explosion as rapidly as possible. To accomplish this, high abort accelerations are required. The design of the acceleration system must consider the human tolerances to accelerations, the distance that the escape body must be displaced from the explosion origin at the time of shock wave intercept, and the overpressure that the capsule structure can withstand.

The highest rate of application of the acceleration (onset rate) encountered will be produced by the escape rocket during the thrust portion of the abort sequence. The propulsion unit (solid propellant rocket) requires 0.020 to 0.025 seconds to build up full thrust after ignition. If the acceleration reaches 15 g at the end of this time, the rate of onset is 750 g/sec; well below the maximum allowable transverse acceleration of 1000 g/sec (Ref 3). The onset rate may be varied by design of the abort rocket so as to extend the time between ignition and full thrust. Therefore, the rate of application of the accelerating force is not a problem.

The direction of the acceleration with respect to the crew is a function of the angle of abort rocket thrust with respect to the longitudinal axis of the abort vehicle, and of the position of the crew. It has been determined (Ref 4) that the optimum tolerance position for transverse acceleration requires the crewman to be leaning forward at an angle of 17 degrees towards the direction of the imposed acceleration. If the thrust of the abort rockets was directly through the longitudinal axis of the APOLLO vehicle this angle would be zero, as the APOLLO crew seats are fixed normal to this axis. However, the selected launch pad abort system (Section 4.1.1.3) utilizes a thrust orientation set 20 degrees from the vertical. By proper positioning of the crew seats relative to the abort vehicle thrust offset, it is possible to achieve an imposed acceleration only 3 degrees from the optimum (Figure III-4-1).

The relatively high drag of the D-2 configuration provides both desirable and undesirable features during the abort acceleration profile. Because of the rapidly increasing drag of the accelerating vehicle, the effective "g" level, seen by the crew during the powered

Ref 3 Handbook of Instructions for Aircraft Designers (HIAD), Air Research and Development Command, ARDCM 80-1, Washington, D.C., 1 July 1959.

Ref 4 Human Tolerance to Prolonged Forward and Backward Acceleration, Neville P. Clark and Stuart Bondurant, WADC Aero Medical Laboratory, Technical Report 59-267, ASTIA Document No. AD 155749, July 1958.

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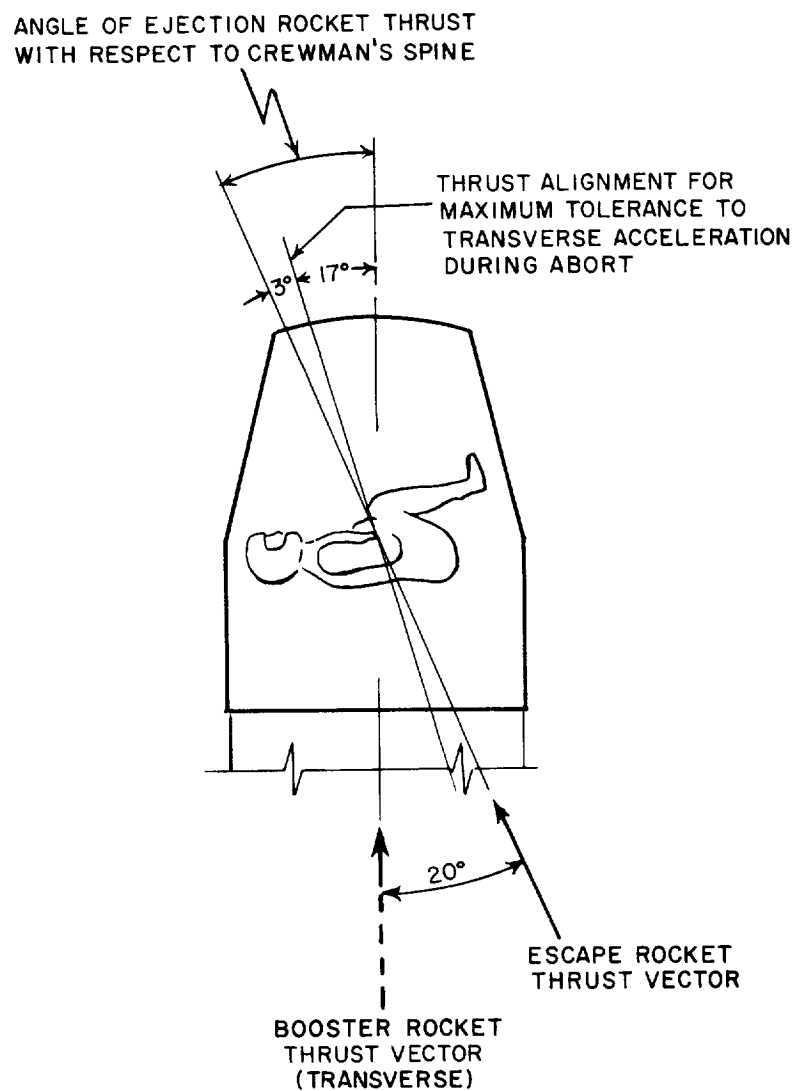


Figure III-4-1. Relationship of escape rocket thrust and crew position for aborts occurring during launch and boost flight

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portion of the abort sequence, diminishes quite rapidly, as shown in Figure III-4-2. However, the high drag causes a relatively high "eyeballs out" acceleration loading to be incurred upon escape rocket burnout. This, plus the acceleration just prior to burnout combine to impose an incremental-g upon the crew which increases with increasing dynamic pressure. (Figure III-4-3)

The R-3 configuration launch pad abort acceleration profile is considerably different due to aerodynamic maneuvering and lower vehicular drag. The initial rocket thrust gives a relatively high axial acceleration and, due to thrust axis inclination of 20 degrees, a small normal acceleration component. As shown in Figure III-4-4, the axial acceleration increases due to decreased vehicular weight with rocket burning, whereas for the D-2 this was offset by the large increase in drag (Figure III-4-2). At the end of rocket burnout a constant-g pullover is performed until, at apogee, the Immelman is completed by a half roll and the vehicle is glided to a landing.

e. Weight Penalty - The weight penalty incurred by the abort rockets is alleviated somewhat by jettisoning rockets at given times during the boost phases. Of the eight abort rockets utilized for the D-2 configuration, four are required during the period from launch through stage S-I burnout only. A 2 percent mission weight penalty is incurred by components carried through 1st stage burnout only, which, for the D-2 vehicle, amounts to less than 20 pounds

f. Compatibility - The launch pad abort system must be compatible with the abort requirements with other mission phases to avoid the necessity for duplication of systems. For both the D-2 and R-3 configurations, the launch abort system is utilized as the primary abort system through the stage S-II burnout of the Saturn vehicle. After this phase the abort system, while not providing the primary abort propulsion, is still utilized to effect separation between the vehicles prior to utilization of the on-board propulsion system.

4.1.1.2 PARAMETRIC STUDIES

In order to weigh the effects of the various limiting criteria previously discussed, and to enable the selection of the desired launch abort system for the D-2 configuration, a series of parametric studies were conducted utilizing the MSVD IBM-7090 computer facility. Presented in Figure III-4-5 are data showing the effects of acceleration on the initial portion of the abort trajectory. Acceleration values given are initial thrust to weight ratios; as the vehicle accelerates the resultant acceleration, due to the essentially constant thrust, decreases. From these data those cases which do not meet the minimum time-displacement requirement for launch pad booster explosion can be rejected.

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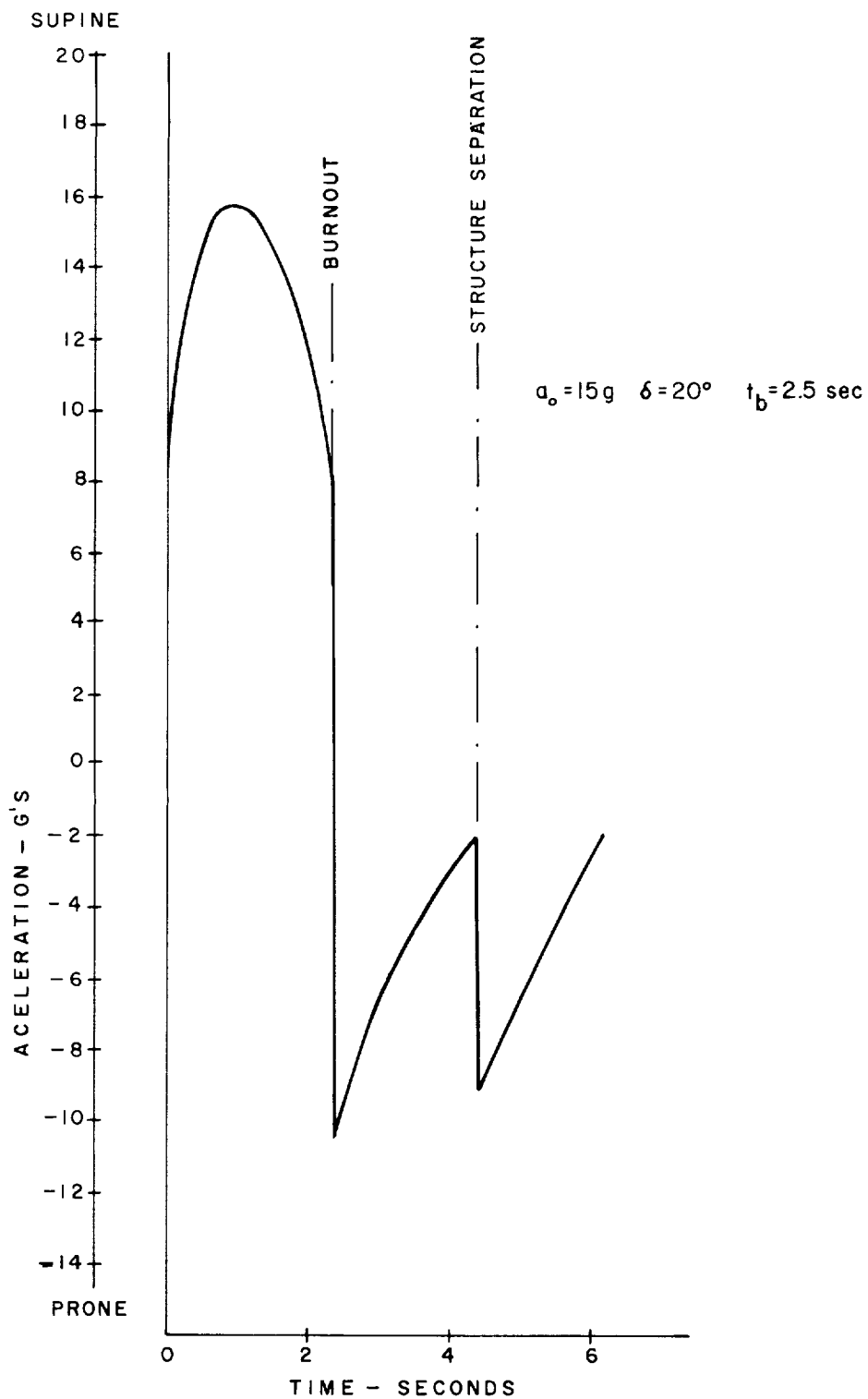


Figure III-4-2. Launch pad abort initial acceleration profile
D-2 Configuration

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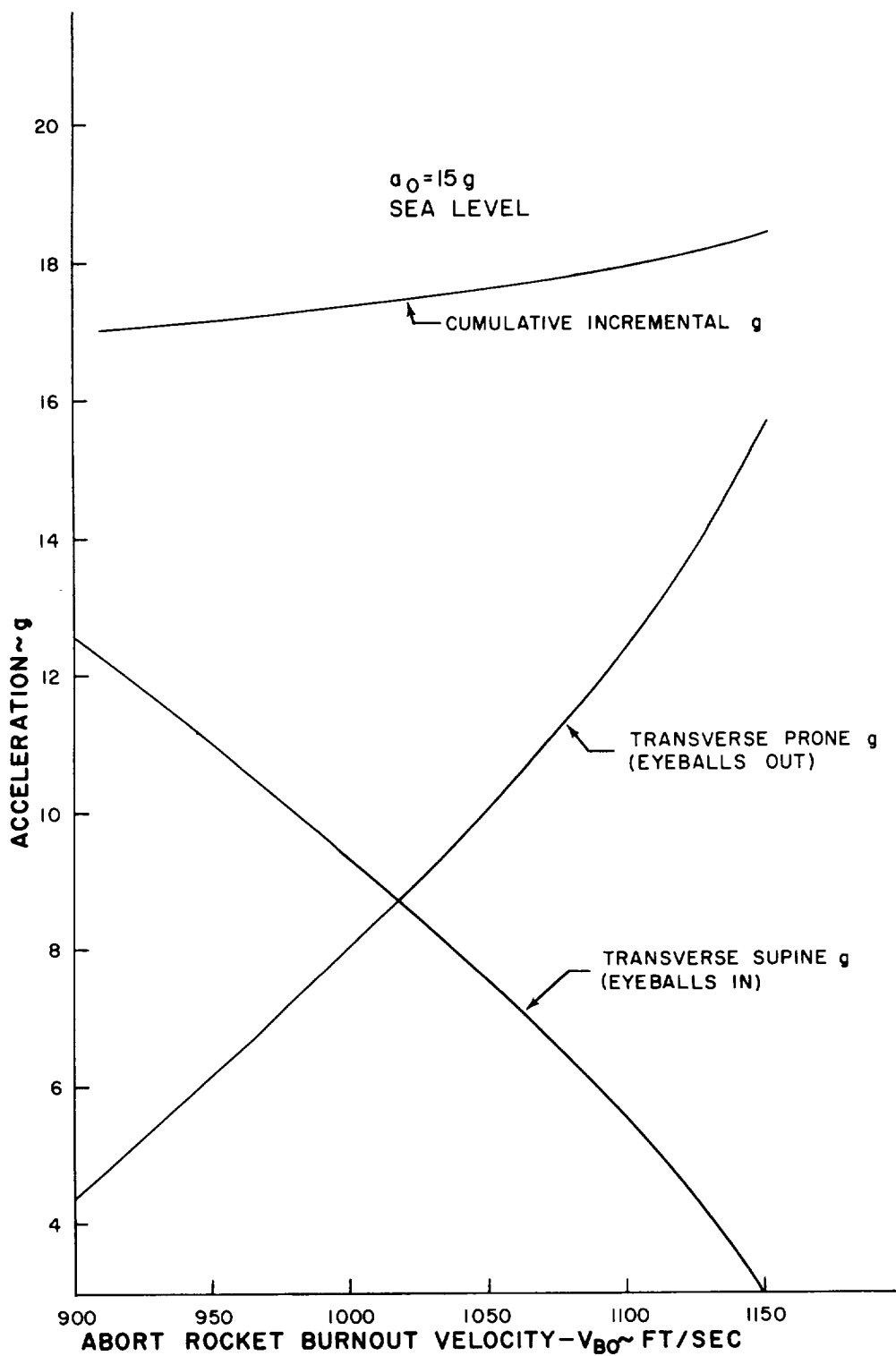


Figure III-4-3. Acceleration at abort rocket burnout
D-2 Configuration

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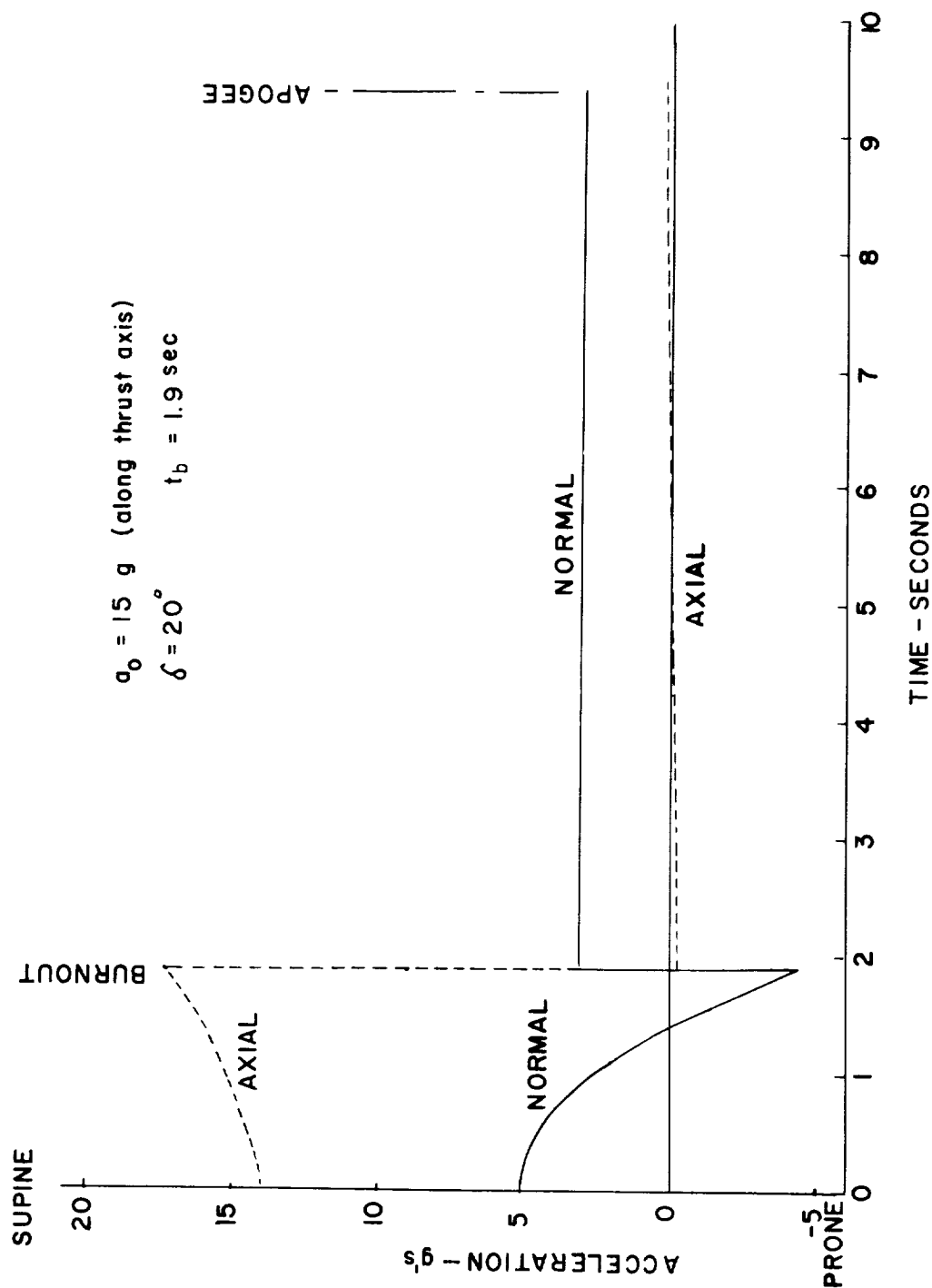


Figure III-4-4. Launch pad abort-initial acceleration profile
D-2 Configuration

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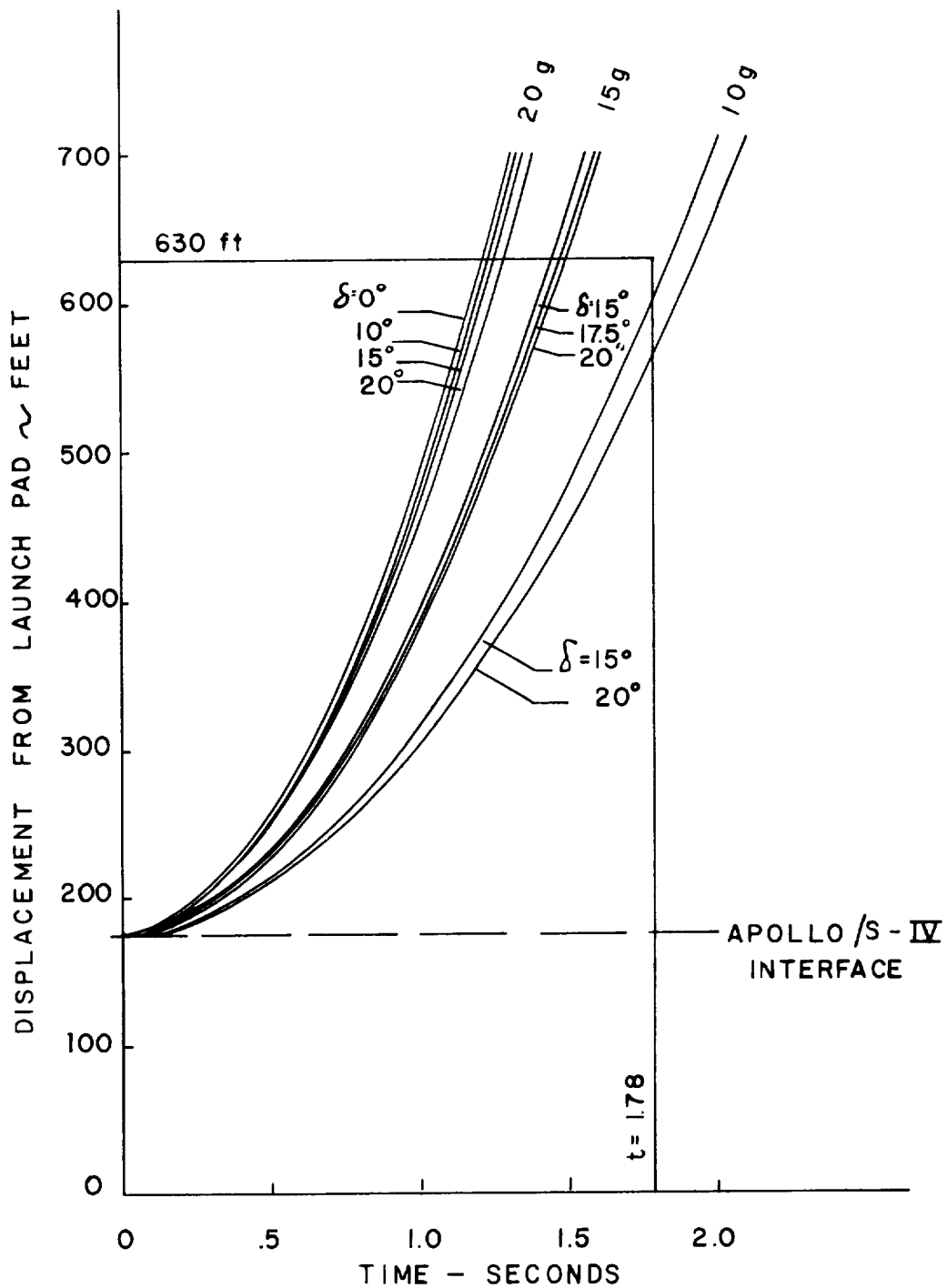


Figure III-4-5. Acceleration required to meet booster explosion criteria of 630 ft. in 1.78 sec.

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In Figure III-4-6 is shown the variation in apogee altitude with abort rocket thrust inclination. The final chute opening altitude, also shown, is a direct function of the apogee altitude as the chute deployment sequence is programmed to start approximately at apogee. Comparison should not be drawn between the 15g and 20g curves as the drag values utilized in the computation were considerably different. Figure III-4-7 shows the variation in range at impact, also as a function of abort rocket thrust inclination. The data presented on these two curves indicates the effect of the thrust inclination.

On the basis of the data from Figures III-4-5 through III-4-7, a series of computer runs were made with the abort rocket burning time and thrust inclination as variables but holding the thrust level, and therefore the initial acceleration, constant. The trajectories for these runs are presented in Figures III-4-8, III-4-9 and III-4-10, showing the effect of thrust axis inclination, and Figures III-4-11, III-4-12 and III-4-13, showing the effect of burning time.

The trade-off and selection of the abort parameters for a manned vehicle cannot be made only on the basis of the data previously presented. The acceleration profile, as shown in Figure III-4-2, must also be weighted and taken into consideration when making a selection. By far the most critical acceleration parameter, for a high drag vehicle such as the D-2, is the acceleration increment which occurs when going from a transverse-supine, or "eyeballs-in", acceleration during rocket boost, to an "eyeballs-out" condition caused by drag immediately after rocket burnout. Presented in Figure III-4-14 are the acceleration values obtained for the D-2 vehicle as a function of rocket burning time. The abscissa of this figure may well have been labeled total impulse, as the acceleration values are a direct function of the burnout velocity which, for relatively small burning times as are considered here, are a function of the total impulse imparted on the abort vehicle. The angle of thrust axis inclination, over the range considered in this study (10 degrees-20 degrees), does not affect the incremental acceleration because of the short burning time.

4.1.1.3 SELECTED SYSTEM

On the basis of the parametric studies of Section 4.1.1.2, a launch pad abort system was selected for the APOLLO D-2 configuration with the following characteristics:

Initial acceleration 15 g

Abort rocket thrust inclination 20 degrees from the vertical

Rocket burning time 2.5 sec.

Apogee altitude 4367 ft.

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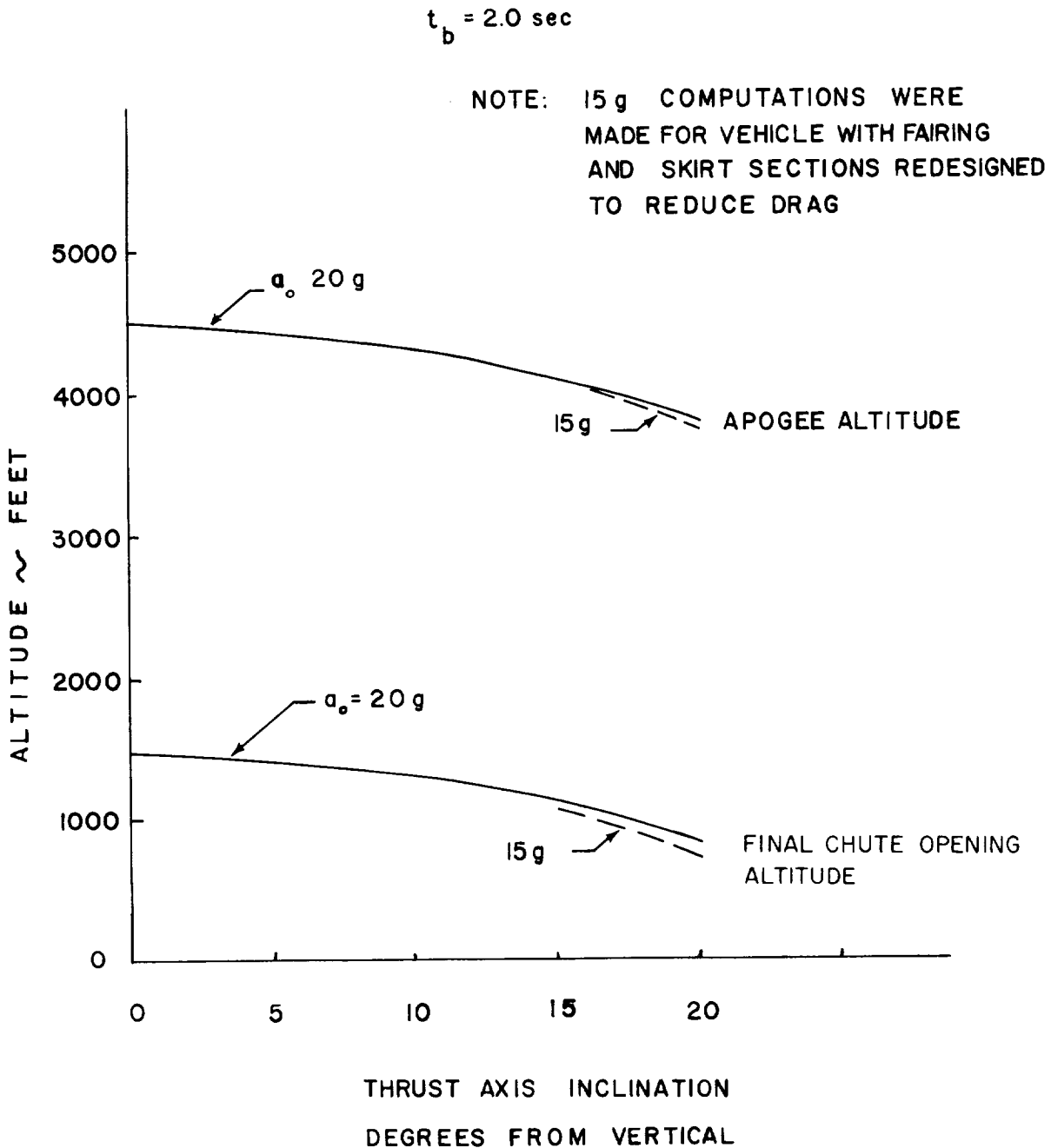


Figure III-4-6. Effect of thrust axis inclination on apogee altitude and final chute deployment altitude — D-2 Configuration

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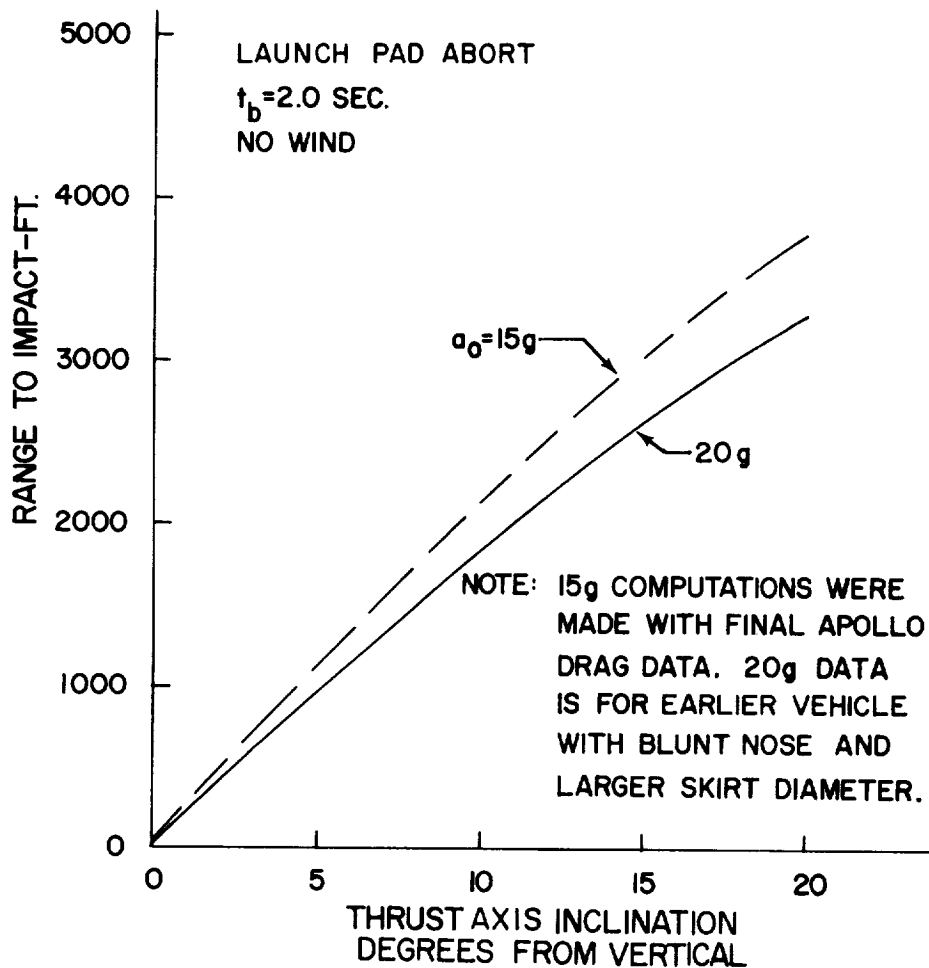


Figure III-4-7. Effect of thrust axis inclination on range to impact
D-2 Configuration

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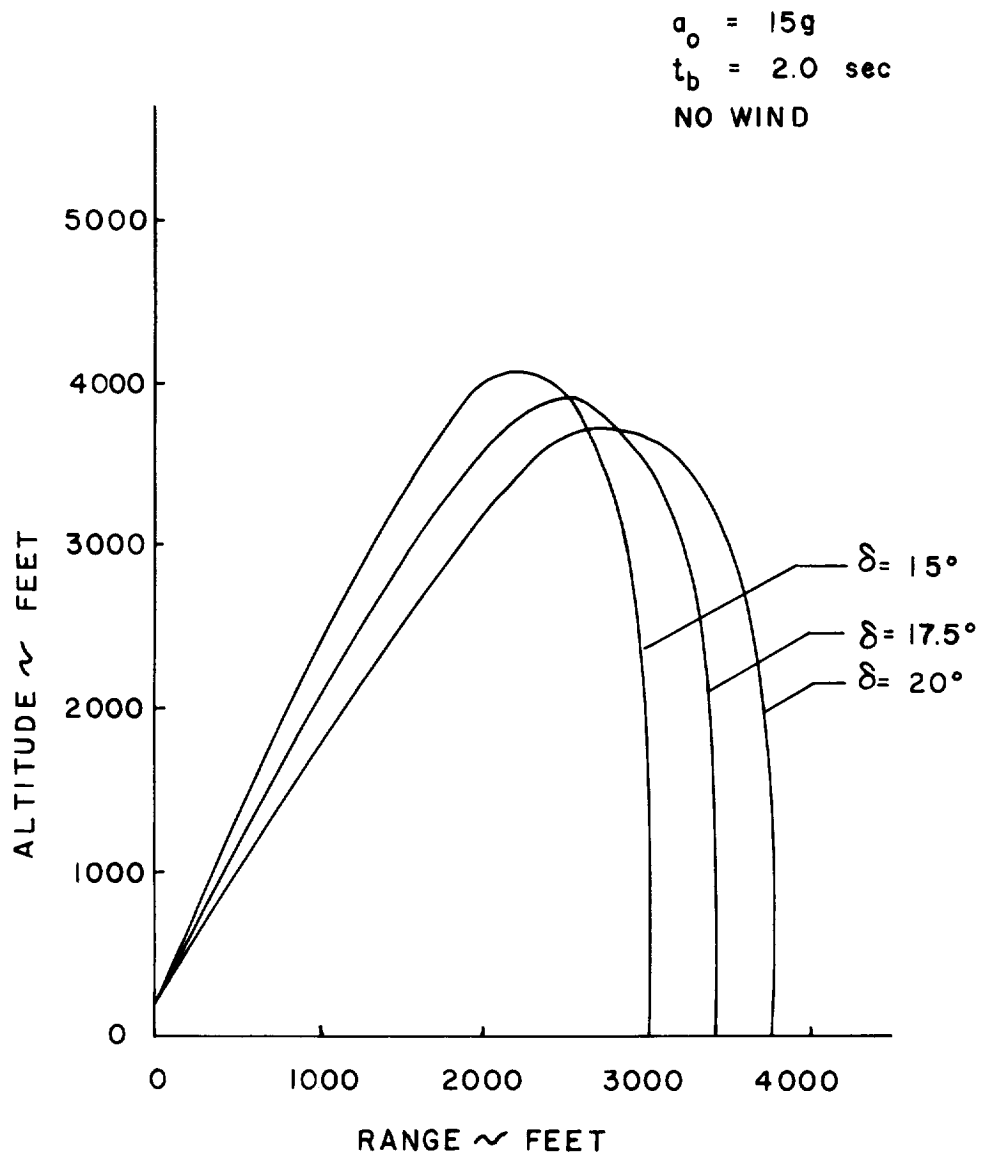


Figure III-4-8. Effect of thrust axis inclination on launch pad abort trajectory
D-2 Configuration

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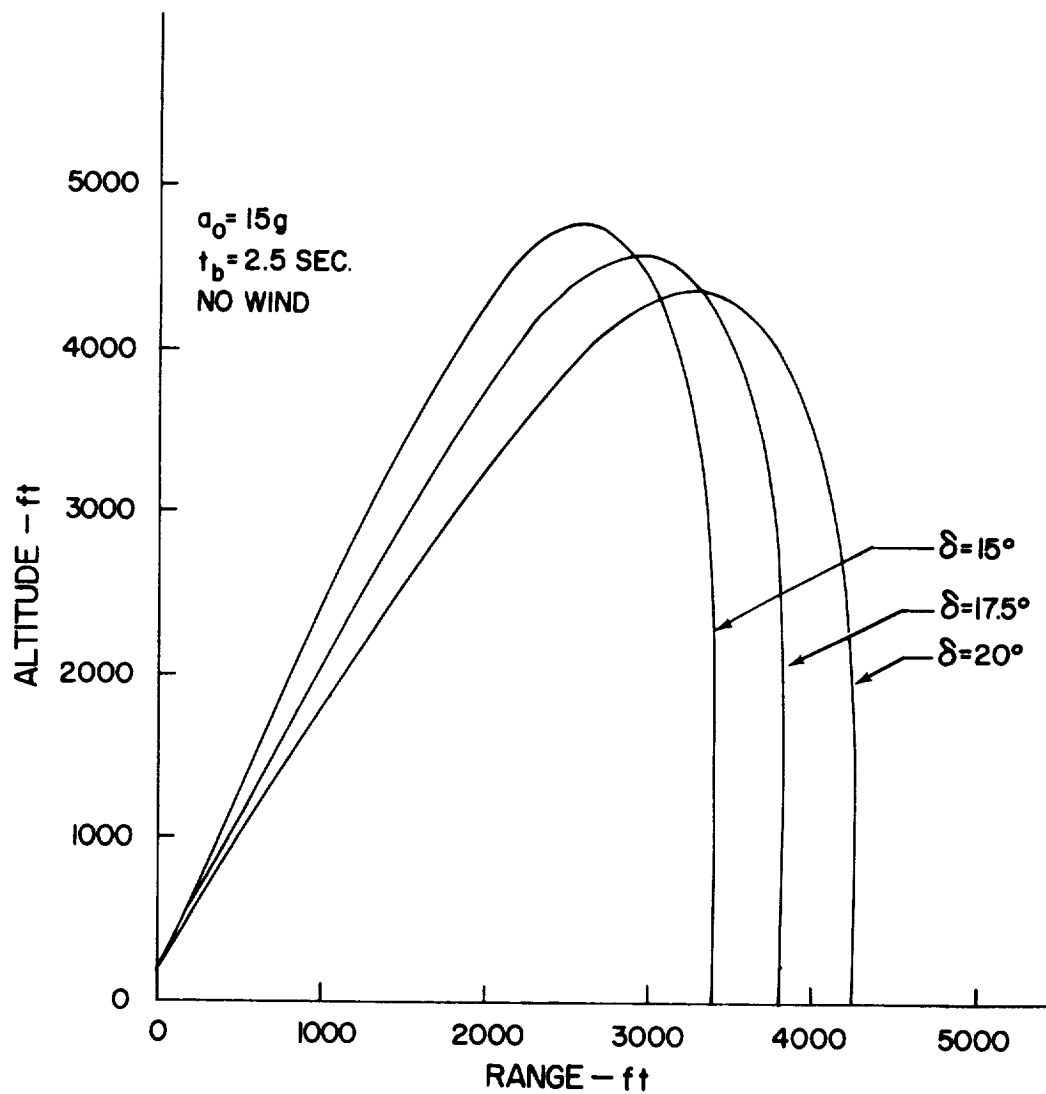


Figure III-4-9. Effect of thrust axis inclination on launch pad abort trajectory
D-2 Configuration

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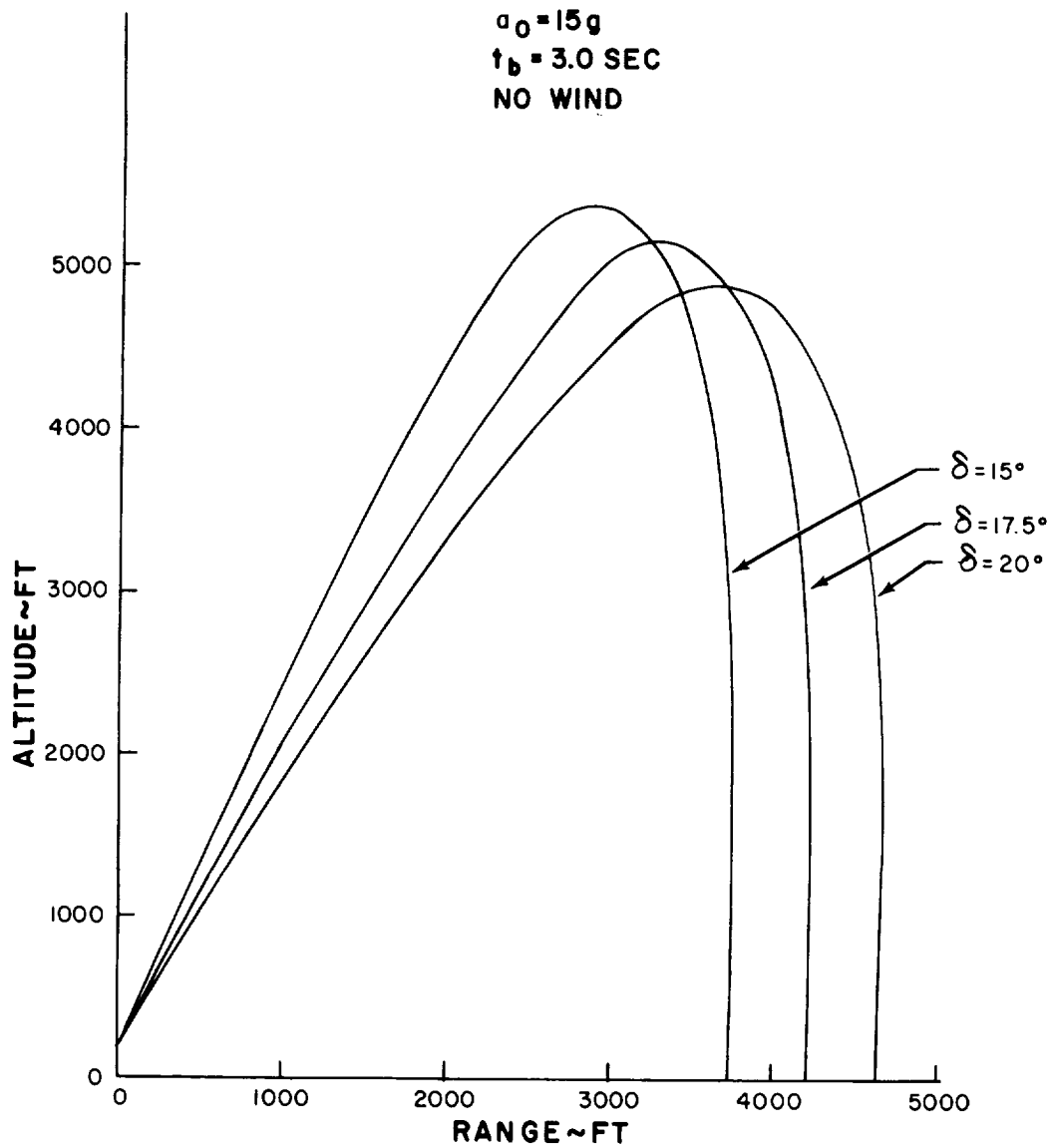


Figure III-4-10. Effect of thrust axis inclination on launch pad abort trajectory
D-2 Configuration

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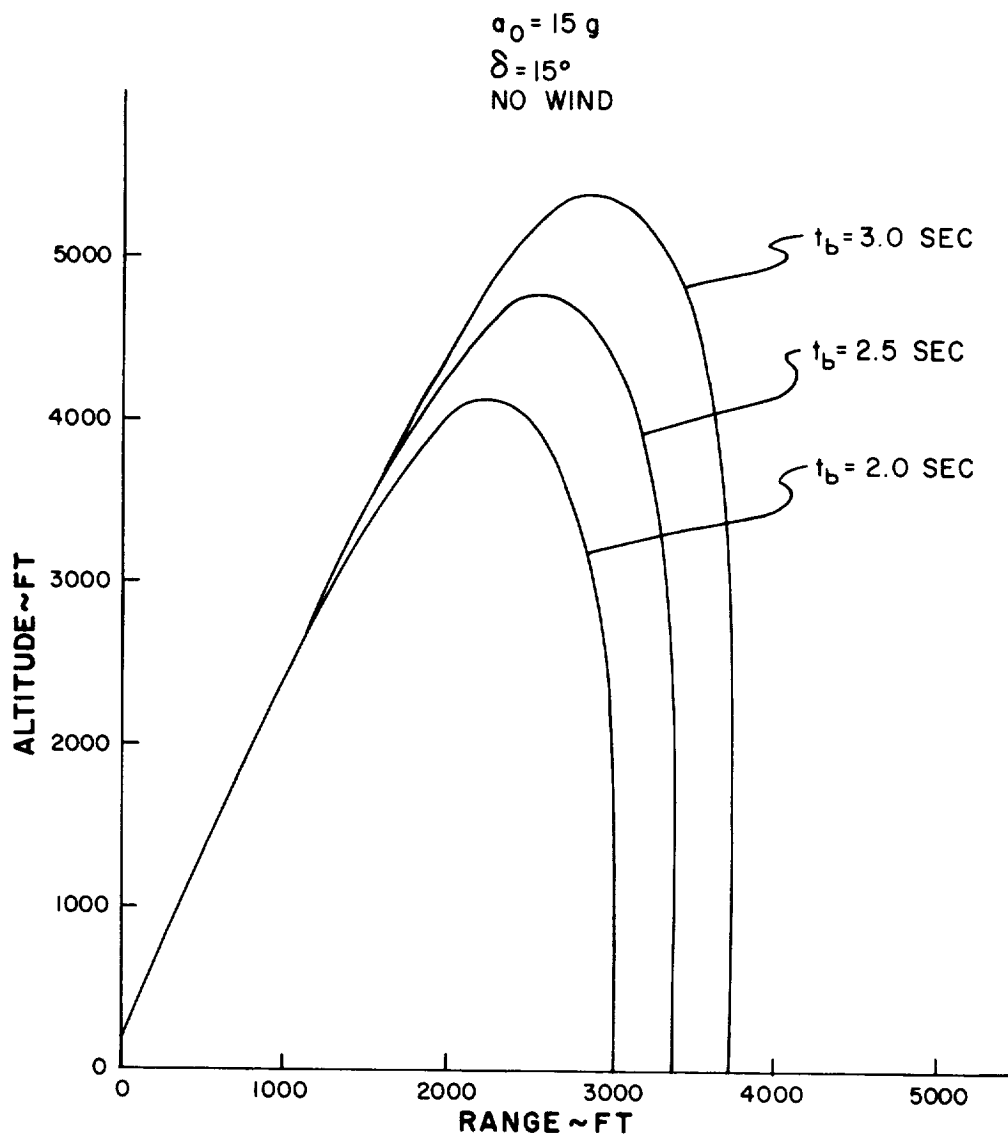


Figure III-4-11. Effect of rocket burning time on launch pad abort trajectory
D-2 Configuration

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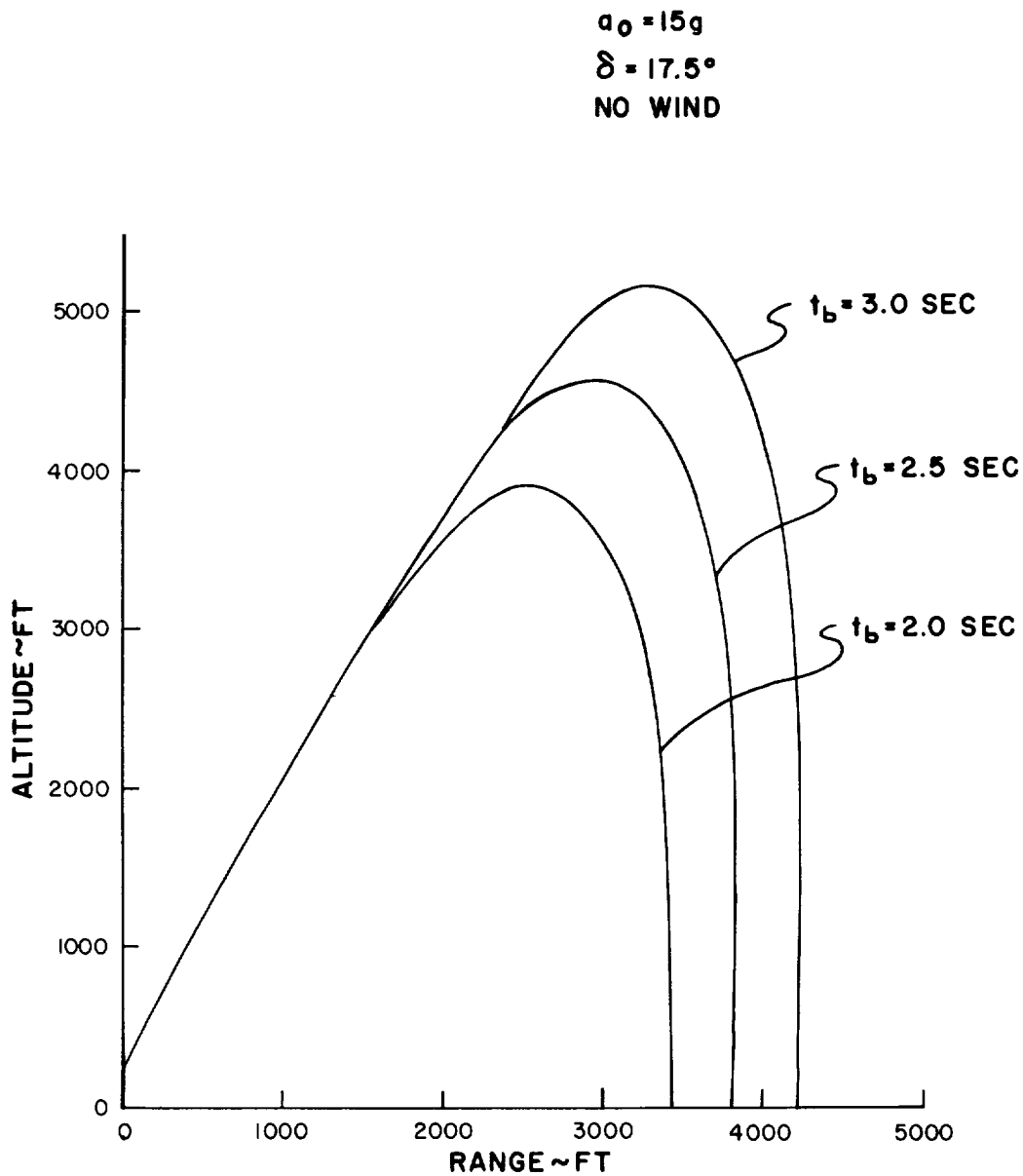


Figure III-4-12. Effect of rocket burning time on launch pad abort trajectory
D-2 Configuration

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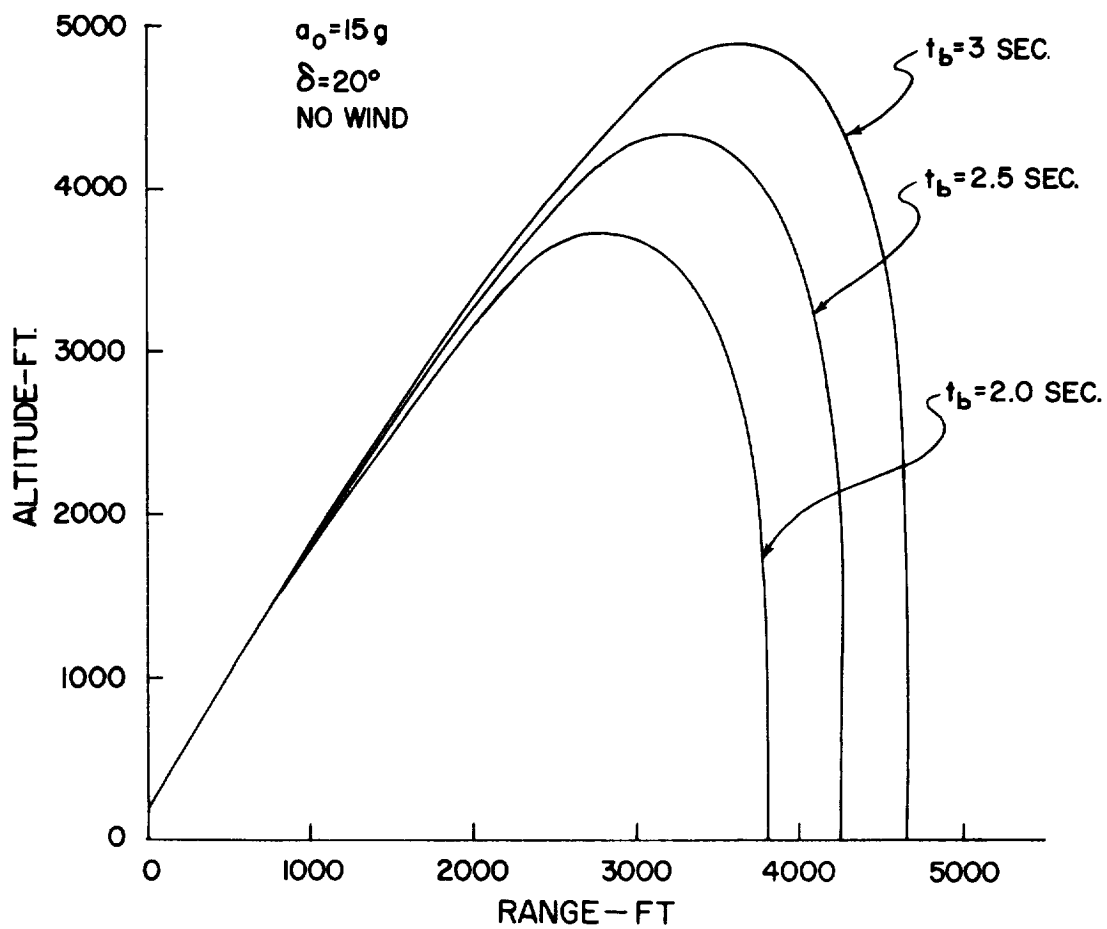


Figure III-4-13. Effect of rocket burning time on launch pad abort trajectory
D-2 Configuration

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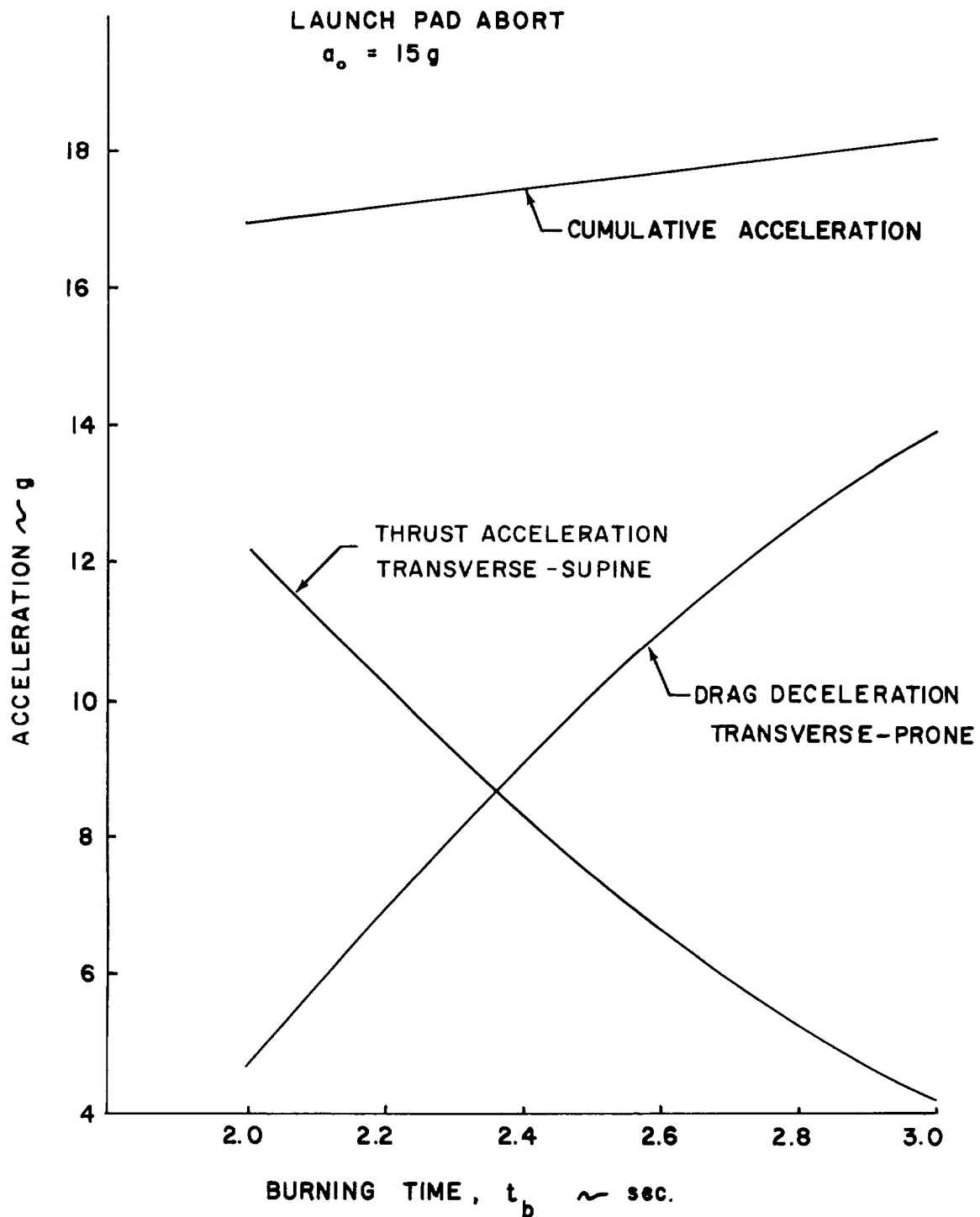


Figure III-4-14. Incremental acceleration at abort rocket burnout
D-2 Configuration

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Velocity at burnout 1052 ft/sec

Altitude of final stage chute deployment 1318 ft.

Range at impact 4240 ft. (no wind)

Acceleration increment 17.6 g

Number of abort rockets 8

Total initial abort rocket thrust 155,600 lbs

The selected trajectory is shown in Figure III-4-15. A time history acceleration profile is shown in Figure III-4-16.

Also considered was the RFP criteria for avoidance of local obstacles. This requires the ability to avoid launch area obstacles such as gantrys, other missiles, and buildings in the event of a launch pad abort with a wind blowing. A comparison of the selected trajectory in zero wind and maximum wind conditions for Saturn launch (28 knots) is shown in Figure III-4-17. Figure III-4-18 shows the possible impact area for the maximum wind from any quarter.

4.1.1.4 SUPPLEMENTAL DATA FOR R-3 CONFIGURATION

To provide the modified lenticular vehicle with an escape capability while on the launch pad and during boost until the dynamic pressure becomes low, six solid propellant rockets are mounted externally on the bottom of the vehicle. The resultant thrust is through the vehicle c.g. at an angle of 20 degrees from the longitudinal axes. For these abort studies the total thrust was taken to be 96,000 pound and the burning time was 1.9 seconds.

Vehicle weight is 6500 pound at abort initiation and 5500 pound after burnout and jettisoning of the rockets.

The criteria which were used for determining abort rocket thrust and burning time were obtained from early NASA data and resulted in a separation distance requirement of 720 feet in 1.82 seconds. These criteria were subsequently reduced to 524 feet in 1.78 seconds as a result of later inputs (see Section 5). The performance which is presented here is, therefore, more than adequate. Subsonic aerodynamic characteristics of the vehicle are given in Volume VI.

Figure III-4-19 presents a time history of the abort in terms of velocity, altitude, and range and Figure III-4-20 shows the axial and normal accelerations. During rocket boost subsonic aerodynamic characteristics for an angle of attack of -20 degrees were used. At 1.82 seconds the vehicle is 740 feet from the top of the Saturn booster which is 170 feet above the pad. Initially the resultant acceleration is 14.77 g's with an axial component of 13.9 g's and a normal component of 5.1 g's. At burnout ($t = 1.9$ seconds) the

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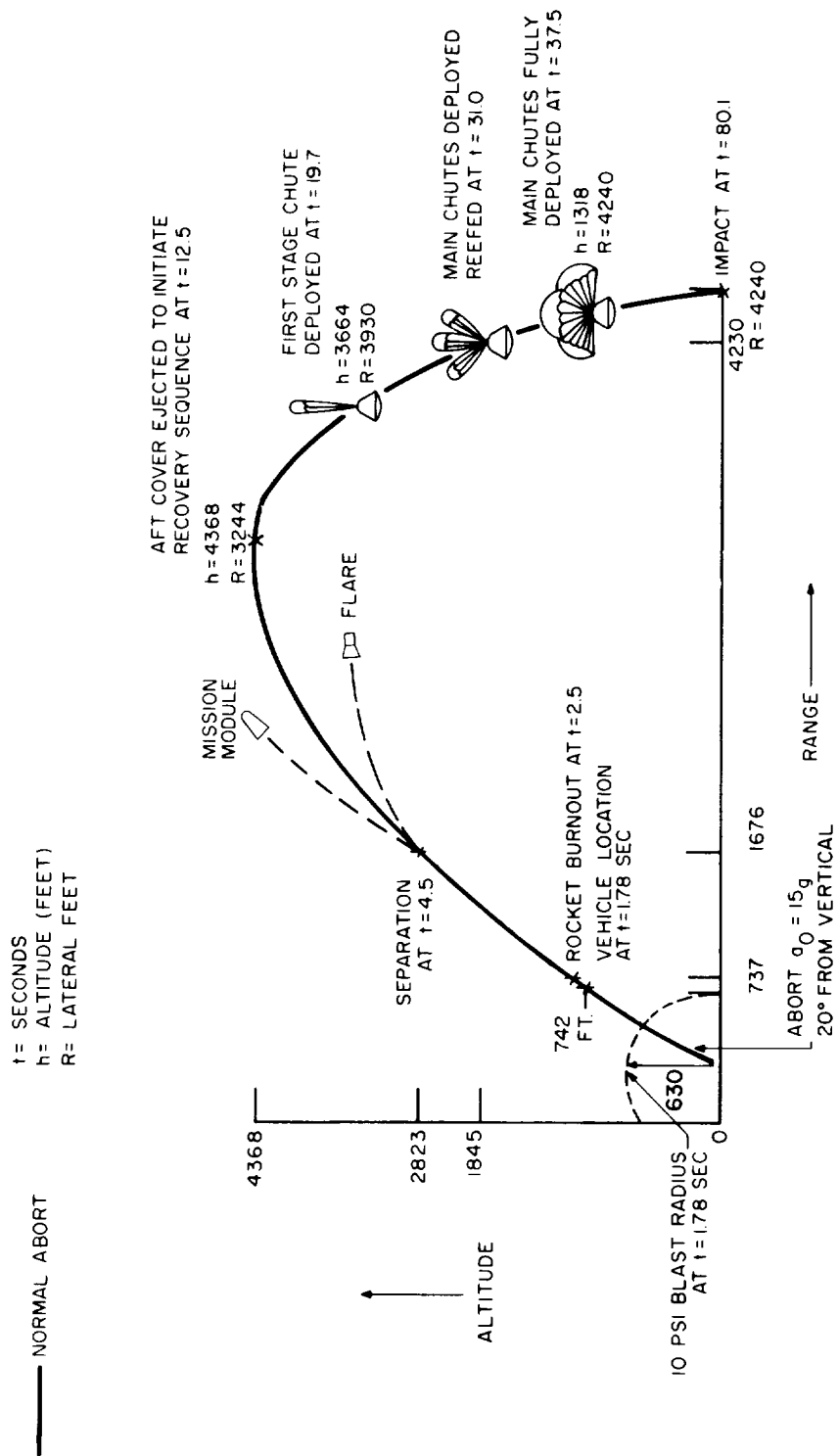
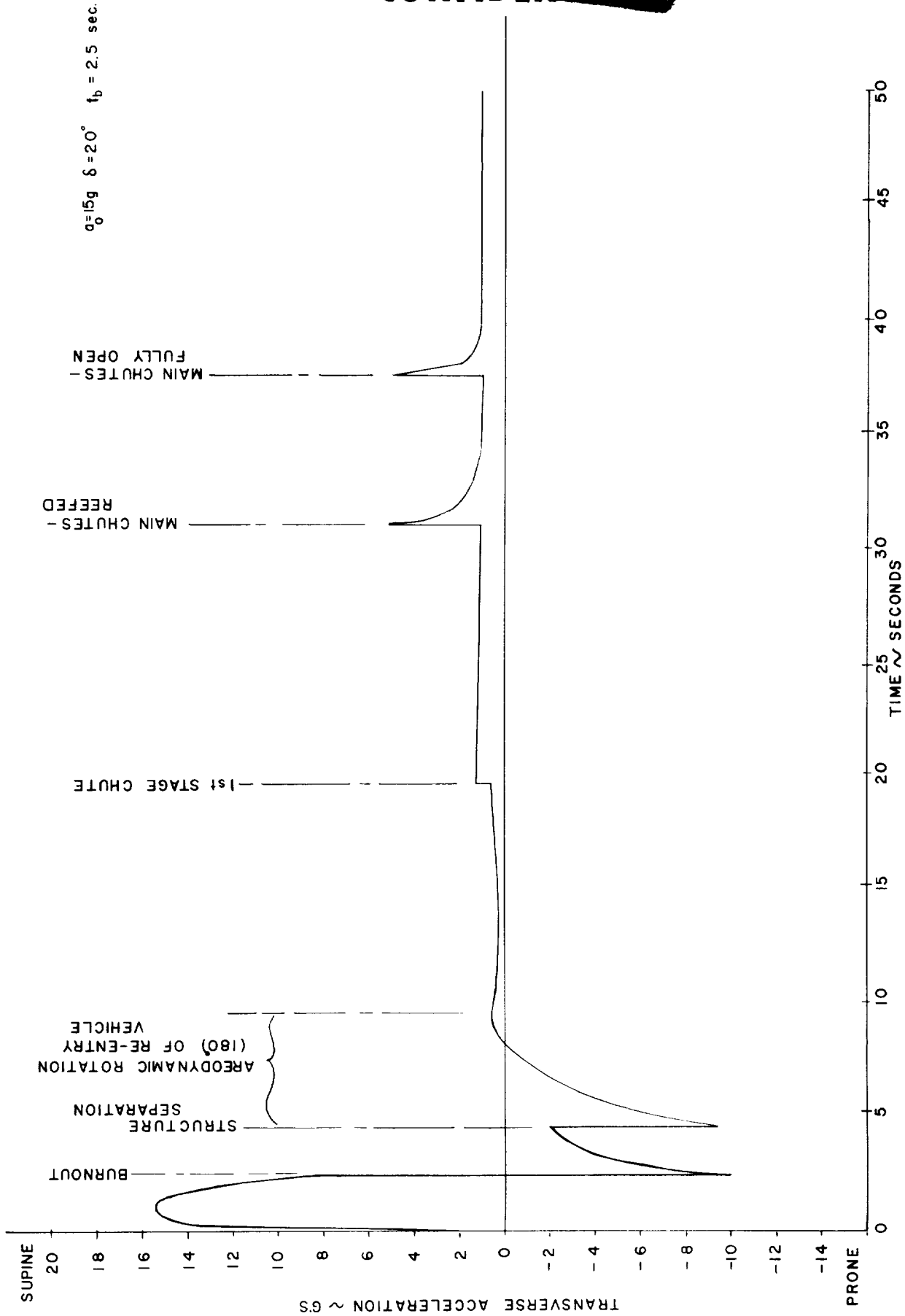


Figure III-4-15. Launch pad abort

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Figure III-4-16. Launch pad abort acceleration profile
D-2 Configuration

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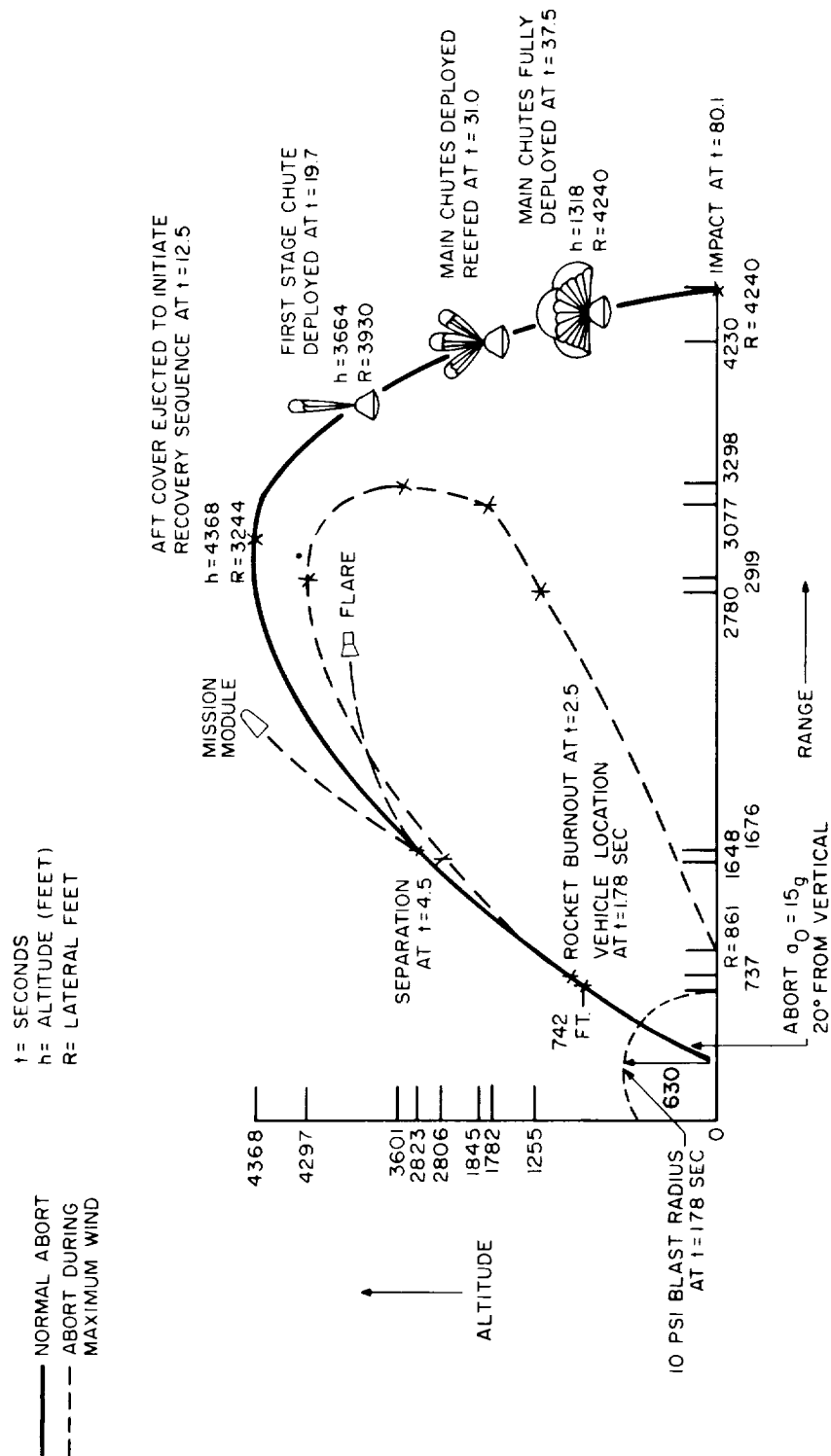


Figure III-4-17. Launch pad abort — effect of wind

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$$\begin{aligned}\delta &= 15^\circ \\ \alpha_0 &= 20g \\ t_b &= 2.5 \text{ SEC}\end{aligned}$$

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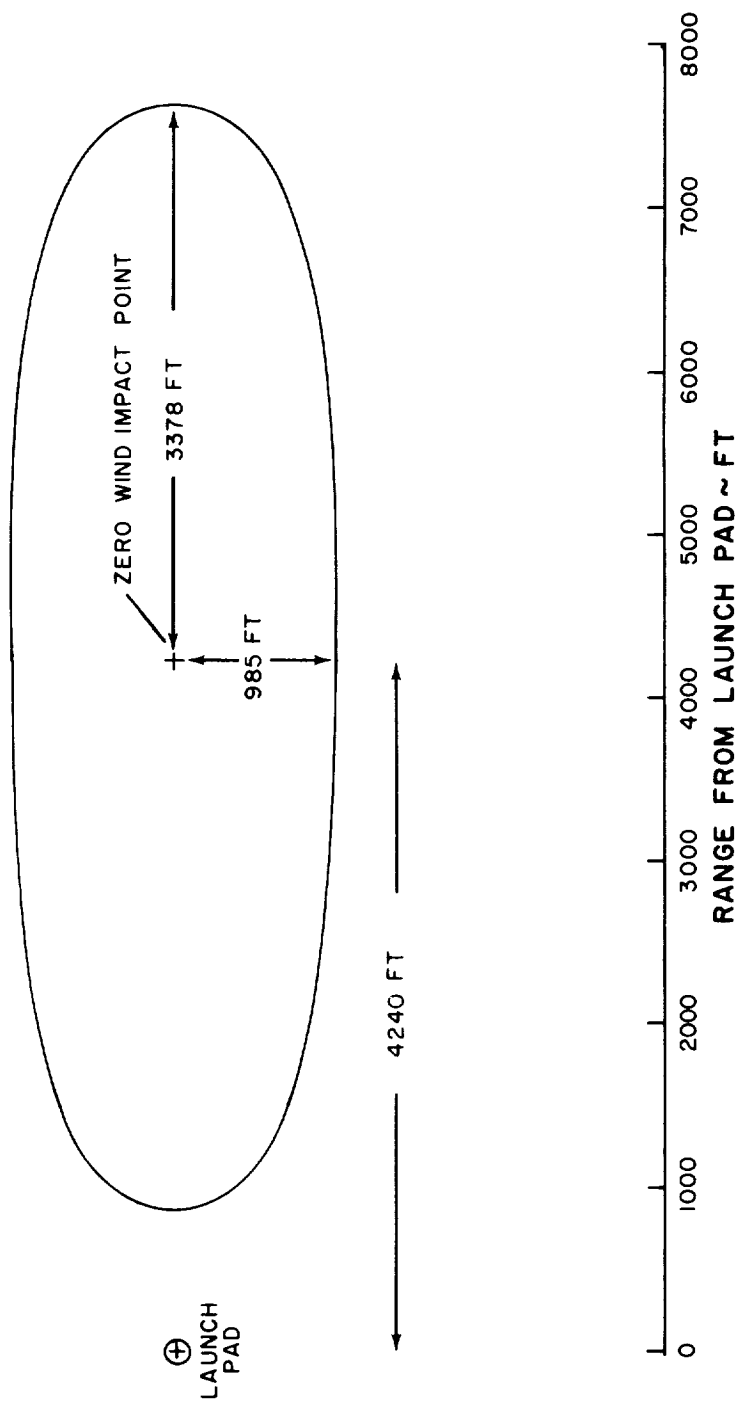
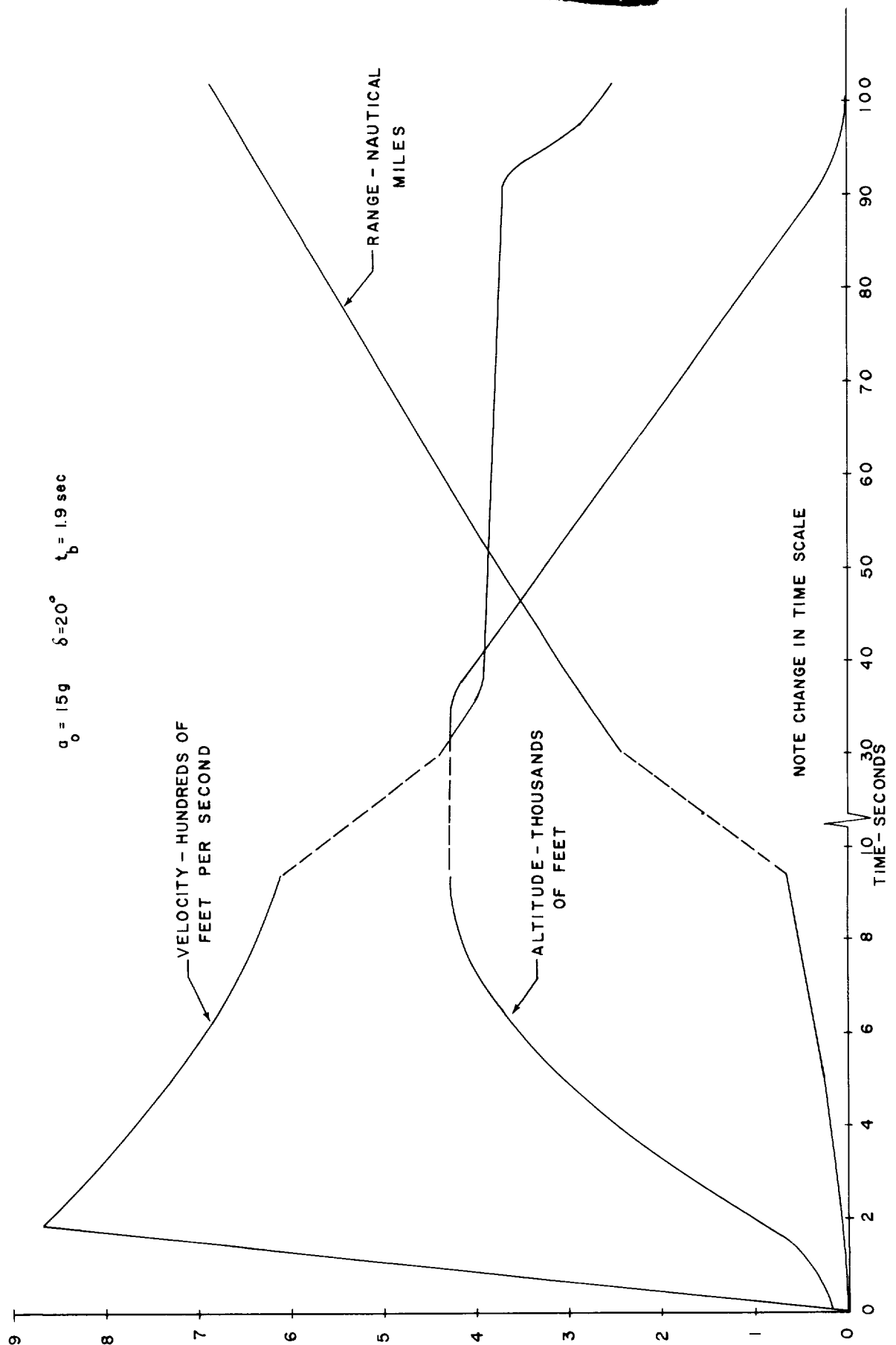


Figure III-4-18. Variation of impact point with wind launch pad abort — D-2 Configuration

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Figure III-4-19. Flight path for abort from launch pad
R-3 Configuration

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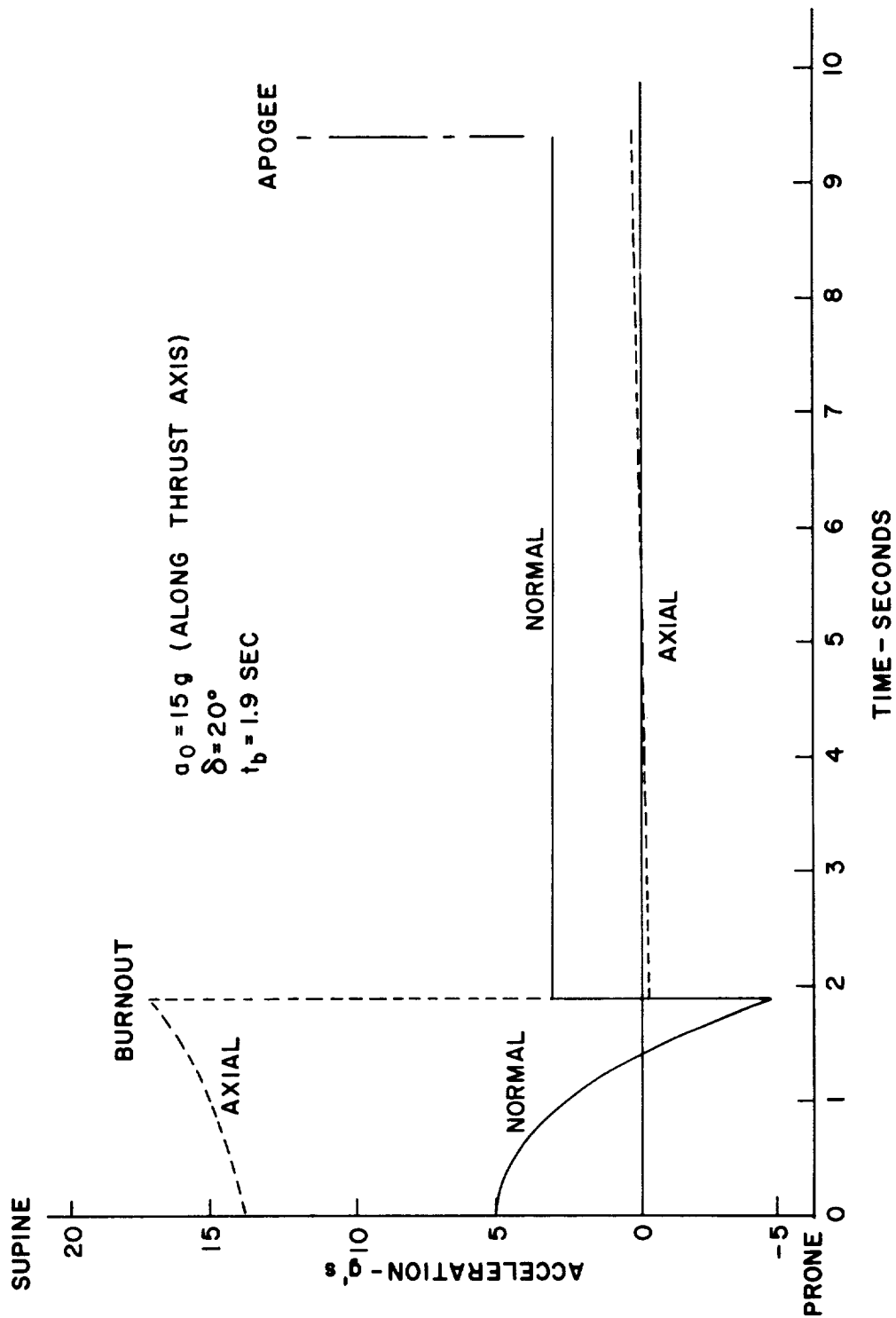


Figure III-4-20. Launch pad abort initial acceleration profile
R-3 configuration

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altitude is 866 feet, the velocity is 910 ft/sec. and the resultant acceleration is 17.9 g's with an axial component of 17.3 g's and a normal component of -4.5 g's. In order to keep the vehicle from passing over the launch pad, an Immelman is performed. This is done by executing a 3 g pull-over until apogee is reached where the vehicle is rolled 180 degrees. The initial change in acceleration due to the pull-over is an increase of 7.7 g's normally and a decrease of 17.5 g's axially. At the top of the maneuver (9.4 seconds after abort initiation) the altitude is 4290 feet and the velocity is 610 ft/sec. After roll-out, level flight is maintained until a steady state descent can be made at $(L/D)_{\max}$. Total range to touchdown is about 7 nmi including a flare at 20 degrees angle of attack. Touchdown velocity is 252 ft/sec.

4.1.2 Abort at Max. q

Although no parametric study, as such, was conducted to determine the optimum system for abort at maximum dynamic pressure, the various thrust-burning time-thrust axis inclination combinations which looked promising during the launch pad abort study, were checked for compatibility with the max. q abort requirements.

4.1.2.1 CRITERIA

For both the D-2 and R-3 configurations the booster explosion criterion, although considerably reduced, still exists at max. q conditions. A complete discussion of these requirements will be found in Section 5. For the relatively high drag D-2 configuration, another equally important factor is the thrust required to overcome the aerodynamic drag, which reaches a maximum at this time. The data (Ref 5) used in these calculations is as follows:

Altitude 32810 ft.

Velocity 1256.6 ft/sec²

Dynamic Pressure 596.99 lb/ft²

Path Angle 27.2 deg.

Time from launch 72 sec.

Range 9843 ft.

Booster Acceleration 1.82 g

Weight 775,000 lbs.

Ref 5 J. T. Markley, Saturn C-1 and C-2 Booster System, NASA Project APOLLO Working Paper No. 1002, Nov. 9, 1960.

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a. Booster Explosion - In order that the maximum overpressure impinging on the abort vehicle, as a result of a booster explosion, be limited to 10 psi, it is necessary to insure that a minimum separation distance of 292 feet be attained within 1.81 seconds. Assuming that the booster continues under normal thrust after the abort sequence is initiated, and indeed actually accelerates more rapidly because of the decrease in weight due to the APOLLO vehicle abort, the abort system selected in section 4.1.1.3 for launch pad abort meets this required criteria.

As developed in Section 5, it is assumed for this study that the actual booster explosion occurs 1.5 seconds after the APOLLO abort vehicle separates from the Saturn. The shock wave from an explosion propagates at transonic velocities equally in all directions from its source and does not partake of the velocity of the vehicle at the time of explosion. As shown in Figure III-4-21, the distance of interest is that between the location of the booster 1.5 seconds after abort and the location of the APOLLO abort vehicle 1.81 seconds after abort.

As a result of the decrease in total booster system weight after abort vehicle separation, the booster, if not shut down, actually accelerates more rapidly

$$a_r = \frac{a_i w_b}{w_b - w_a} \quad (1)$$

where a_r = resultant booster acceleration

a_i = booster system acceleration prior to abort

w_b = booster system weight prior to abort

w_a = weight of abort vehicle

$$a_r = \frac{1.82 \times 32.2 \times 775,000}{(775,000 - 9,445)} = 59.3 \text{ ft/sec}^2 \quad (2)$$

The average booster velocity in the ensuing 1.81 seconds is:

$$v_r = v_i + \frac{a_r \Delta t}{2} \quad (3)$$

where v_i = initial booster system velocity

v_r = resultant average booster velocity

Δt = time to explosion

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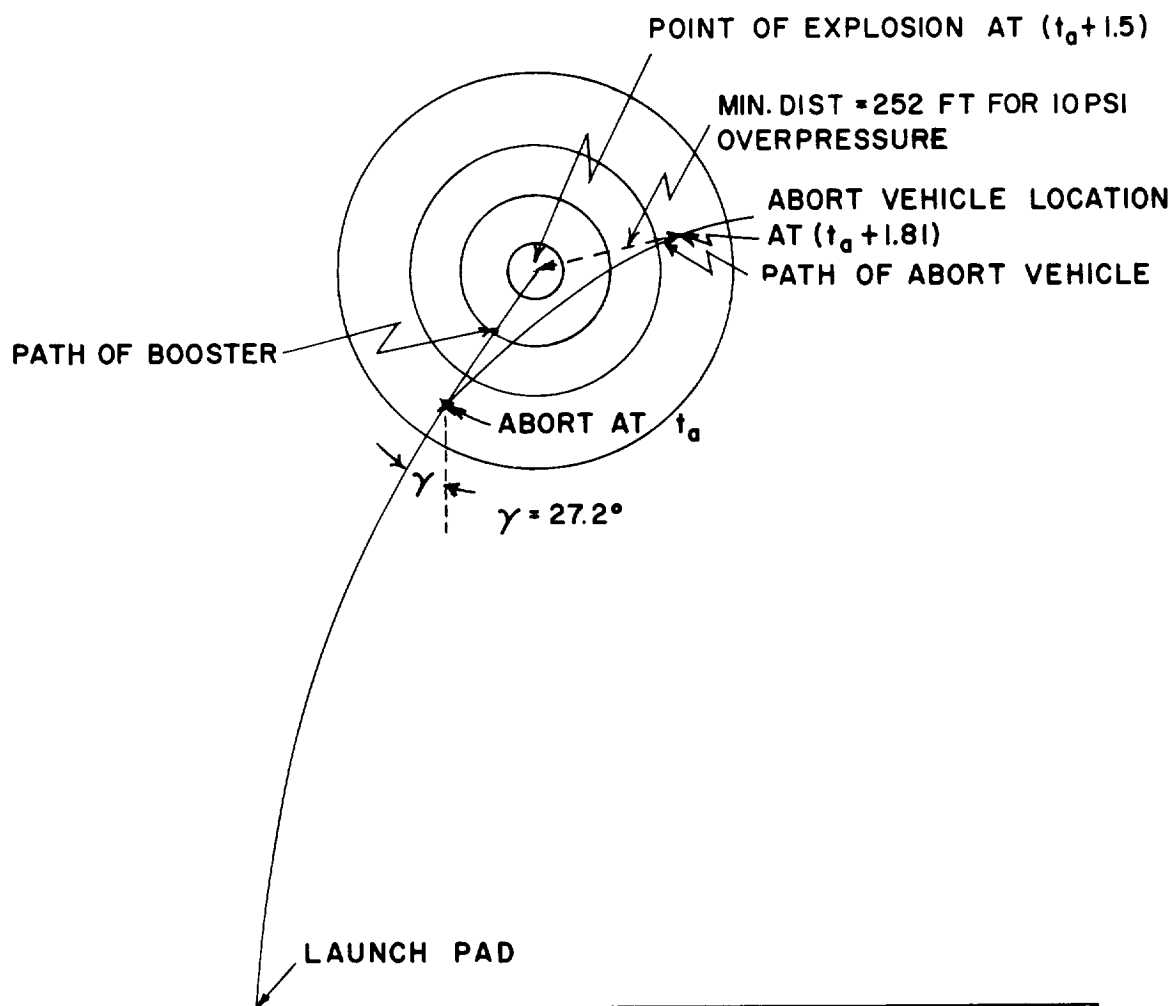


Figure III-4-21. Flight path relationships at max.q abort

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$$v_r = 1256.6 + \frac{59.3 \times 1.5}{2} = 1301.1 \text{ ft/sec} \quad (4)$$

Therefore, the distance travelled by the booster from abort initiation to the point of explosion is

$$d_r = 1301.1 \times 1.5 = 1952 \text{ ft.} \quad (5)$$

As the booster is inclined to the vertical 27.2 degrees, the distance travelled, in rectangular coordinates

$$\Delta h_b = d_r \cos \gamma = 1736 \text{ ft} \quad (6)$$

$$\Delta s_b = d_r \sin \gamma = 892 \text{ ft} \quad (7)$$

The abort vehicle is accelerated by the launch pad abort rocket system with a thrust axis offset of 20 degrees. At the end of 1.81 seconds it has reached a position of

$$h_a = 35,067 \text{ ft} \quad (8)$$

$$s_a = 11,189 \text{ ft} \quad (9)$$

or has traveled from the start of abort

$$\Delta h_a = 2257 \text{ ft} \quad (10)$$

$$\Delta s_a = 1346 \text{ ft} \quad (11)$$

with the abort vehicle so oriented that the abort rocket thrust axis is 20 degrees down from the booster longitudinal axis. The resultant distance that the abort vehicle attains prior to being overtaken by the 10 psi shockwave is

$$R = \sqrt{(h_a - h_b)^2 + (s_a - s_b)^2} = 691 \text{ ft} \quad (12)$$

b. Aerodynamic Drag - Although the same thrust is applied during max q abort as for launch pad abort, the acceleration of the vehicle is considerably retarded due to aerodynamic drag. The acceleration is however, as shown in the preceding section, sufficient to provide more than adequate separation between the booster and the abort vehicle.

c. Recovery System Operation - The same recovery system sequences are utilized at max q, for both the D-2 and R-3 configurations, as were for the launch pad abort. For the D-2 a separation initiated timer starts the parachute deployment sequence at 12.5 seconds. At this time the vehicle is at 42425 feet with a dynamic pressure of approximately 40 lb/sq ft. Apogee is reached approximately 6.5 seconds later at 43140 feet altitude. During the intervening period very little additional drag is encountered as

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the first stage First ribbon chute is not fully open until 7.2 seconds after initiation of the sequence. Final stage chute deployment occurs at 40,247 feet. Descent time from this point is slightly in excess of 16 minutes.

d. Acceleration Profile - The time history acceleration profile for the max q abort case is shown in Figure III-4-22. Although the initial acceleration is fairly low due to aerodynamic drag, the acceleration increment between powered and ballistic flight remains approximately the same as for launch pad abort.

4.1.2.2 SELECTED SYSTEM PERFORMANCE

The trajectory of the selected system for max q abort is shown in Figure III-4-23. The characteristics, utilizing the same boost system as for the launch pad abort, are as follows:

| | |
|--|-------------------|
| Initial acceleration | 8.4 g |
| Abort rocket thrust inclination | 20° from vertical |
| Rocket burning time | 2.5 sec. |
| Number of abort rockets | 8 |
| Total initial abort rocket thrust | 155,600 lbs. |
| Apogee altitude | 43140 ft |
| Velocity at burnout | 1784 fps |
| Altitude of final stage chute deployment | 40,247 |
| Range at impact | 21,274 (no wind) |
| Acceleration increment | 17.5 g |

4.1.2.3 SUPPLEMENTARY DATA FOR R-3 CONFIGURATION

For the R-3 lenticular configuration the max q abort maneuver is similar to that utilized in the launch pad abort case. As for the D-2 configuration, the criteria for separation between vehicle and booster from this point is 292 feet in 1.81 seconds, assuming the booster continues its normal trajectory. Aerodynamic characteristics for the modified lenticular vehicle were estimated from the data of Reference 6.

Figure III-4-24 shows the trajectory of this vehicle during abort rocket burning and the subsequent Immelman relative to the booster ascent trajectory. Figure III-4-25 is a time history of axial and normal accelerations through the maneuver. While the abort rockets are burning the vehicle is trimmed at $C_L = 0$ to give a low drag flight. The initial normal acceleration is 5 g's and the axial acceleration is 10.9 g's, the resultant being 12 g's. At 1.81 seconds the vehicle is about 700 feet from the booster. At burnout the velocity is 1818 ft/sec, the altitude is 35,550 feet and the flight path angle is 74.2 degrees. Axial acceleration has decreased to 9.3 g's and normal acceleration has increased to 5.6 giving a resultant of 10.9 g's.

Ref 6 NASA unpublished data; "Project APOLLO, Summary of Studies of a Lenticular Vehicle Capable of Entry from a Lunar Mission," NASA STG, Langley Field, Va., March 3, 1961.

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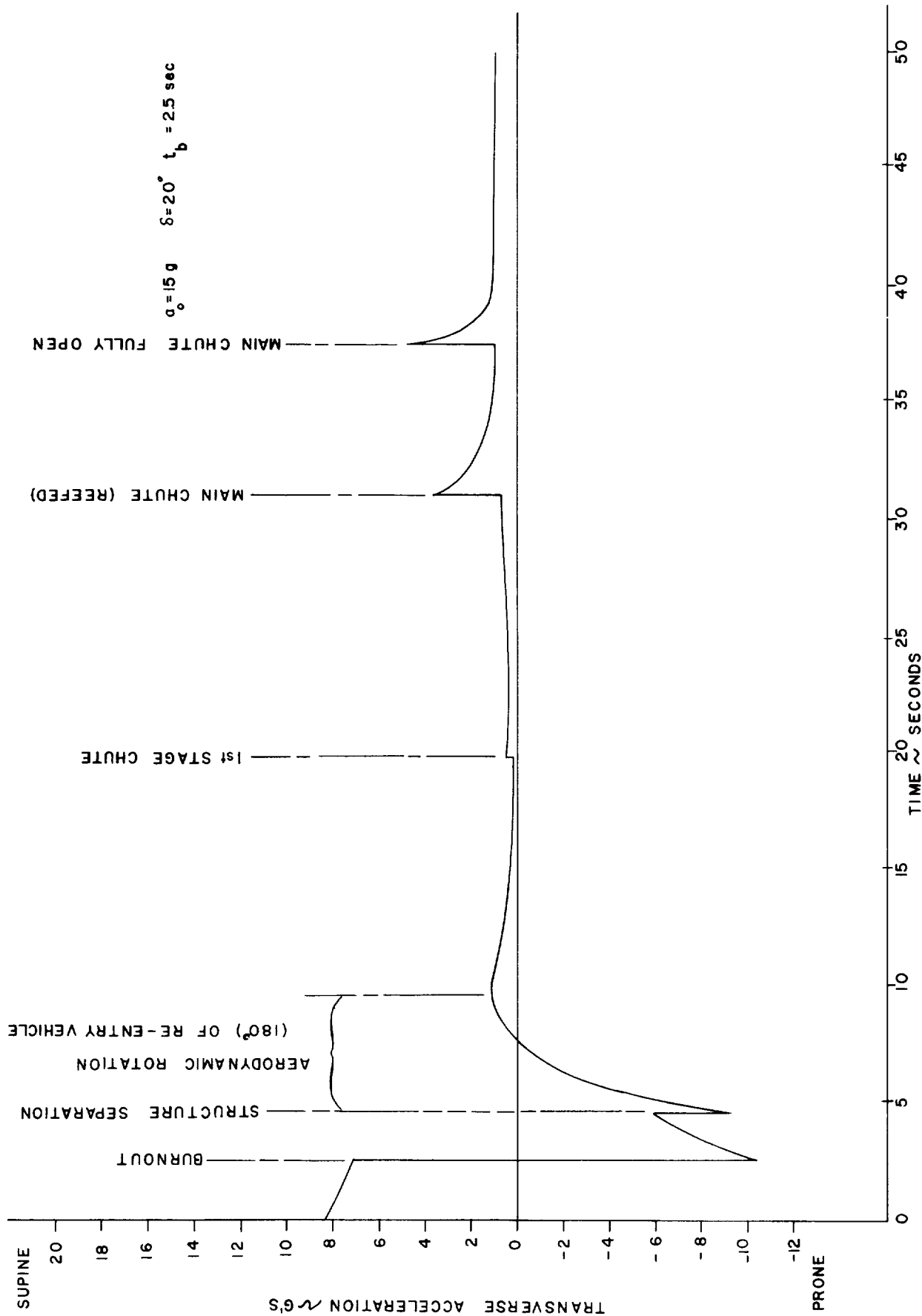


Figure III-4-22. Max. q abort acceleration profile
D-2 Configuration

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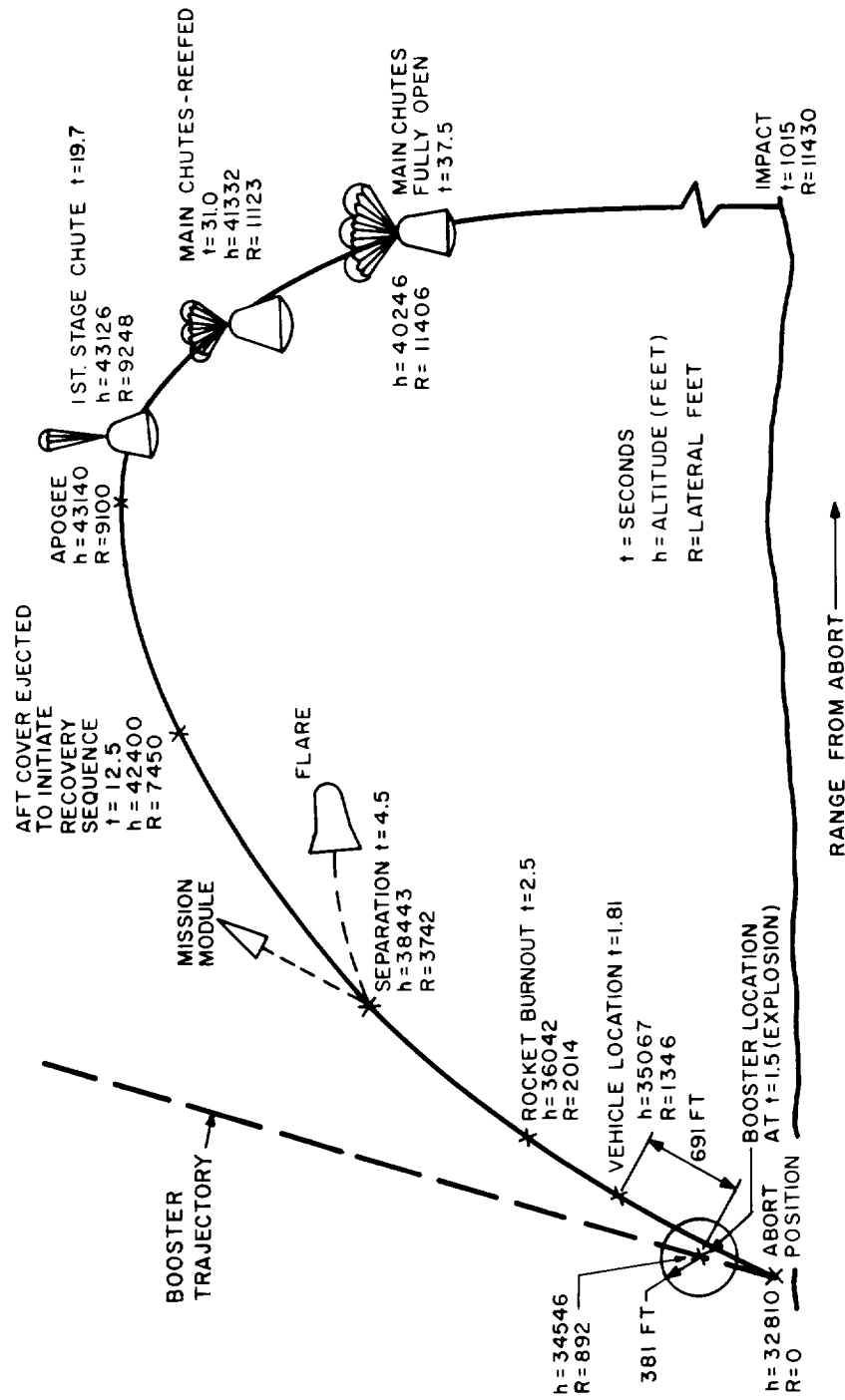


Figure III-4-23. Max. q abort profile D-2 Configuration

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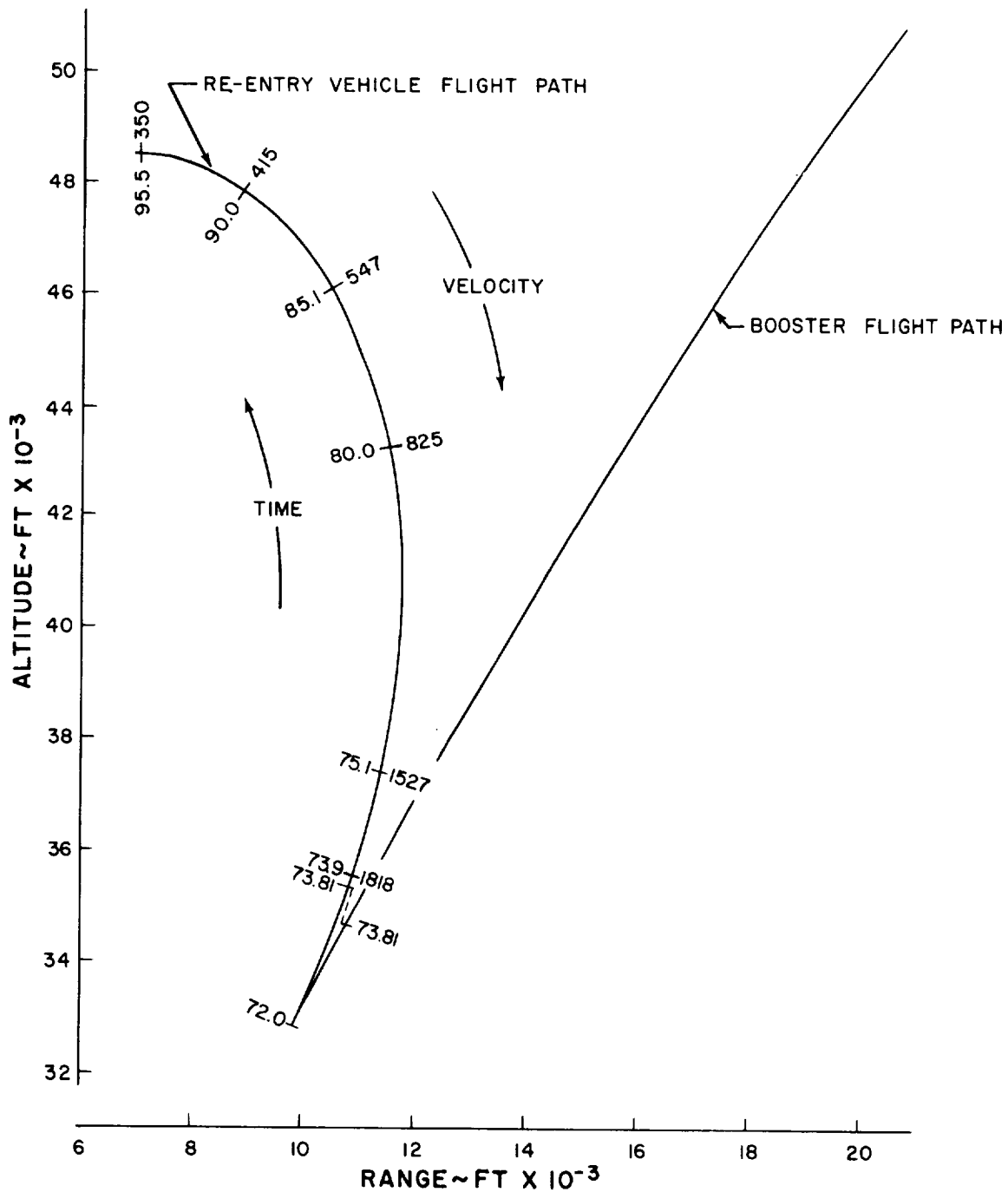


Figure III-4-24. Abort from high q conditions
R-3 Configuration

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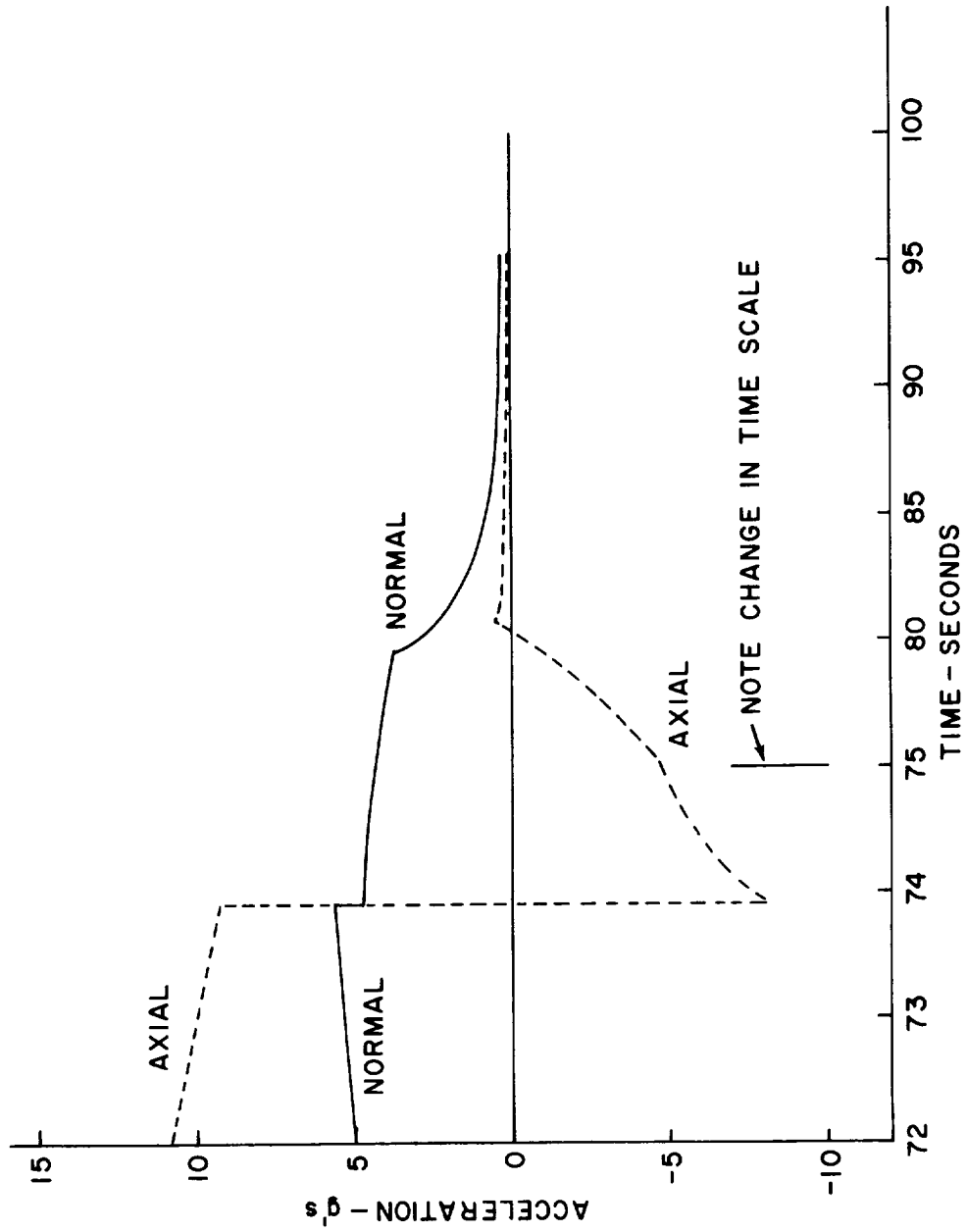


Figure III-4-25. Time history of accelerations during max. q abort
R-3 Configuration

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A 3-g pull-over is then initiated which reverses the axial acceleration to -8.2g's, a total change of 17.5 g's. The normal acceleration is decreased by 1 g. At transonic speeds, the ability to pull normal g's is limited by C_{LMAX} to values of approximately 1.5. This decreases as velocity decreases. The maneuver (except for rolling out) is completed at 48,580 feet altitude (apogee) with a velocity of 350 ft/sec. Resultant flight is in the direction of the launch pad at a distance of 1.1 nmi downrange.

4.1.3 Stage S-I Burnout

Just prior to burnout the Saturn S-I stage reaches its maximum acceleration. However, with the launch pad abort system still available, the capabilities of the system greatly exceed the thrust requirements due to booster acceleration or aerodynamic drag.

4.1.3.1 CRITERIA

The booster explosion problem is no longer applicable. The explosion shock wave propagation velocity from the point of explosion is transonic and does not include the velocity of the booster at the time of explosion. Therefore the abort vehicle, and the components of the booster itself, will easily outrun the shock wave as the vehicle is traveling at $M = 2.45$ at the time. The data (Ref 5) used for calculations at this stage are as follows:

| | |
|----------------------|---------------------------|
| Altitude | 72182 ft |
| Velocity | 2368.9 ft/sec |
| Dynamic Pressure | 370.89 lb/ft ² |
| Path Angle | 43.1 deg. |
| Time from launch | 98.2 sec |
| Range | 36091 ft |
| Booster Acceleration | 2.54 g |
| Weight | 621,000 lb |

a. Aerodynamic Drag - With reduced dynamic pressure and a lower drag coefficient due to high Mach numbers, the abort system, still utilizing the eight launch pad abort rockets, has sufficient thrust to easily overcome the drag.

b. Recovery System Operation - Eighty seconds after launch (approximately 45,000 ft) the abort programmer/sequencer disarms the 12.5 second abort timer and arms the normal recovery system sensors. These are a baroswitch set for 25,000 feet with a g-switch initiated timer as backup. The re-entry vehicle will descend in free ballistic flight after separation until these devices, or the crewmembers, actuate the recovery sequence.

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c. Acceleration Profile - The abort acceleration at, or just prior to Saturn S-I stage burnout is shown in Figure III-4-26. Even though the deceleration due to drag is reduced extensively at abort rocket burnout, the total acceleration gradient at this point remains relatively the same as for the conditions previously considered due to the higher acceleration at burnout. Because of the relatively low apogee of the abort profile the maximum "re-entry" acceleration is only 1.6 g.

4.1.3.2 D-2 SELECTED SYSTEM PERFORMANCE

The trajectory of the abort vehicle for conditions of abort at Saturn S-I burnout is presented in Figure III-4-27. The characteristics of this profile, utilizing the launch pad abort rocket system, are as follows:

| | |
|---------------------------------------|-----------------------|
| Initial Acceleration: | 12.8 g |
| Burnout Acceleration: | 14.5 g |
| Abort rocket thrust inclination: | 20° from vertical |
| Rocket burning time: | 2.5 sec. |
| Number of abort rockets: | 8 |
| Total Initial abort rocket thrust: | 155,600 lbs |
| Apogee altitude: | 114,388 ft. |
| Velocity at burnout: | 3368 fps |
| Chute deployment initiation altitude: | 25,000 ft. |
| Range at 25,000 ft: | 198,719 ft. (no wind) |
| Acceleration increment: | 17.6 g |

4.1.4 Stage S-II Ignition

At Saturn stage S-I burnout, four of the eight abort rockets are jettisoned from the APOLLO vehicle. The four remaining rockets have sufficient thrust available to propel the abort vehicle away from the booster throughout S-II burning.

4.1.4.1 CRITERIA

In essence the only difference between the abort under this condition and that of Section 4.1.3, stage S-I burnout, is the reduction in Saturn acceleration and abort propulsion. The abort acceleration profile just after Saturn Stage S-II ignition is shown in Figure III-4-28.

4.1.4.2 SELECTED SYSTEM PERFORMANCE

The trajectory for abort at Stage S-II ignition is presented in Figure III-4-29 with characteristics as follows:

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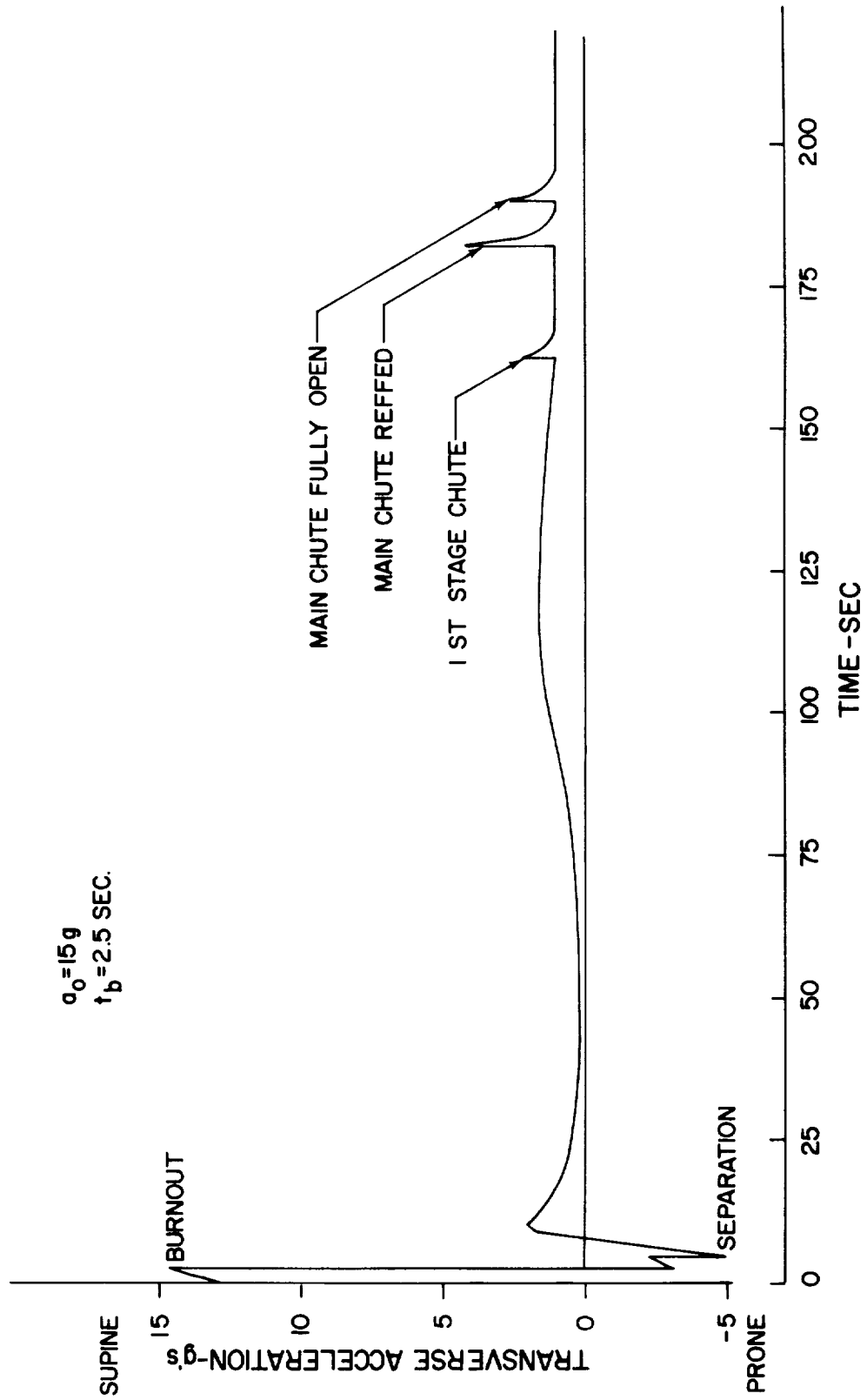


Figure III-4-26. Abort acceleration profile at Saturn S-I burnout
D-2 Configuration

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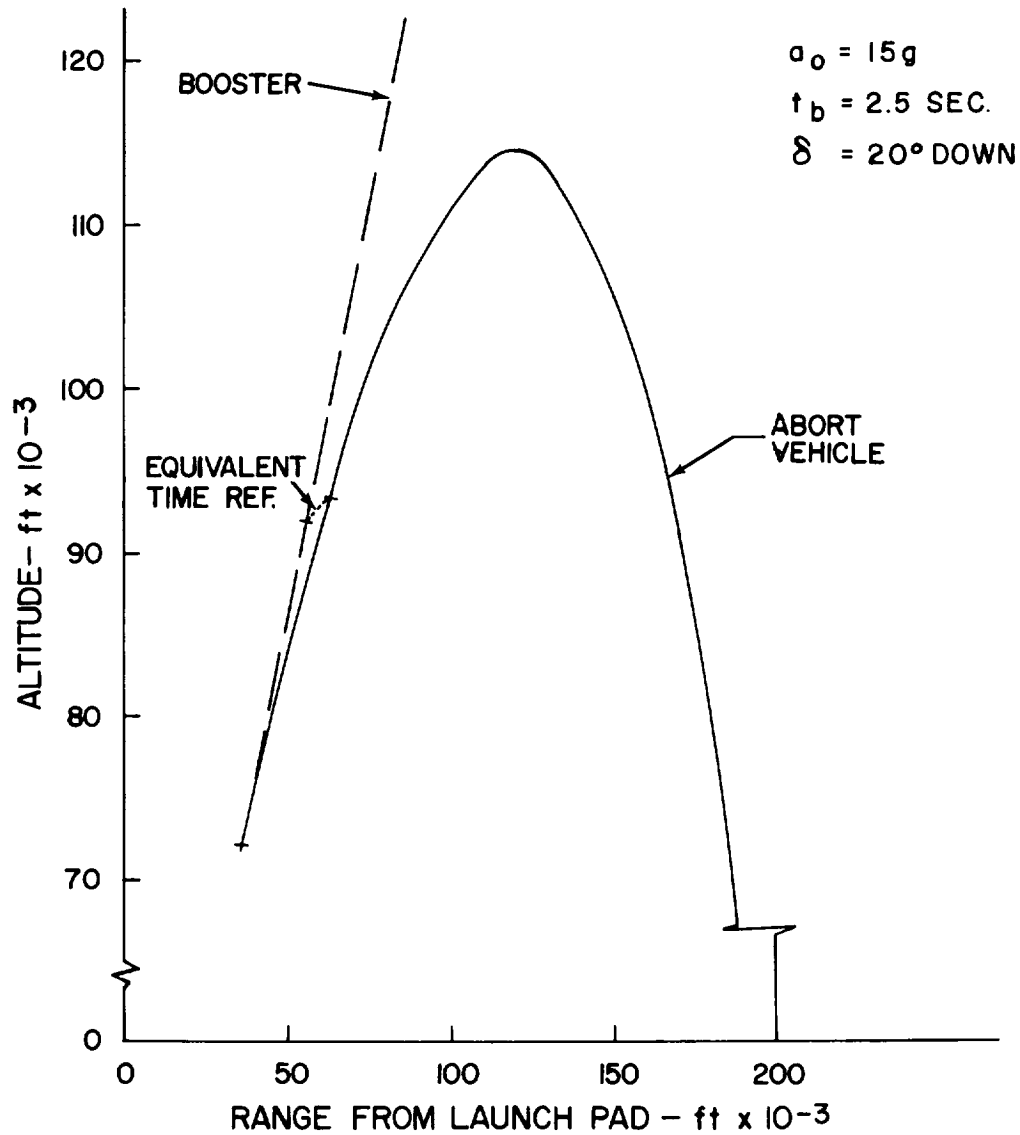


Figure III-4-27. Abort trajectory at Saturn S-I burnout
D-2 Configuration

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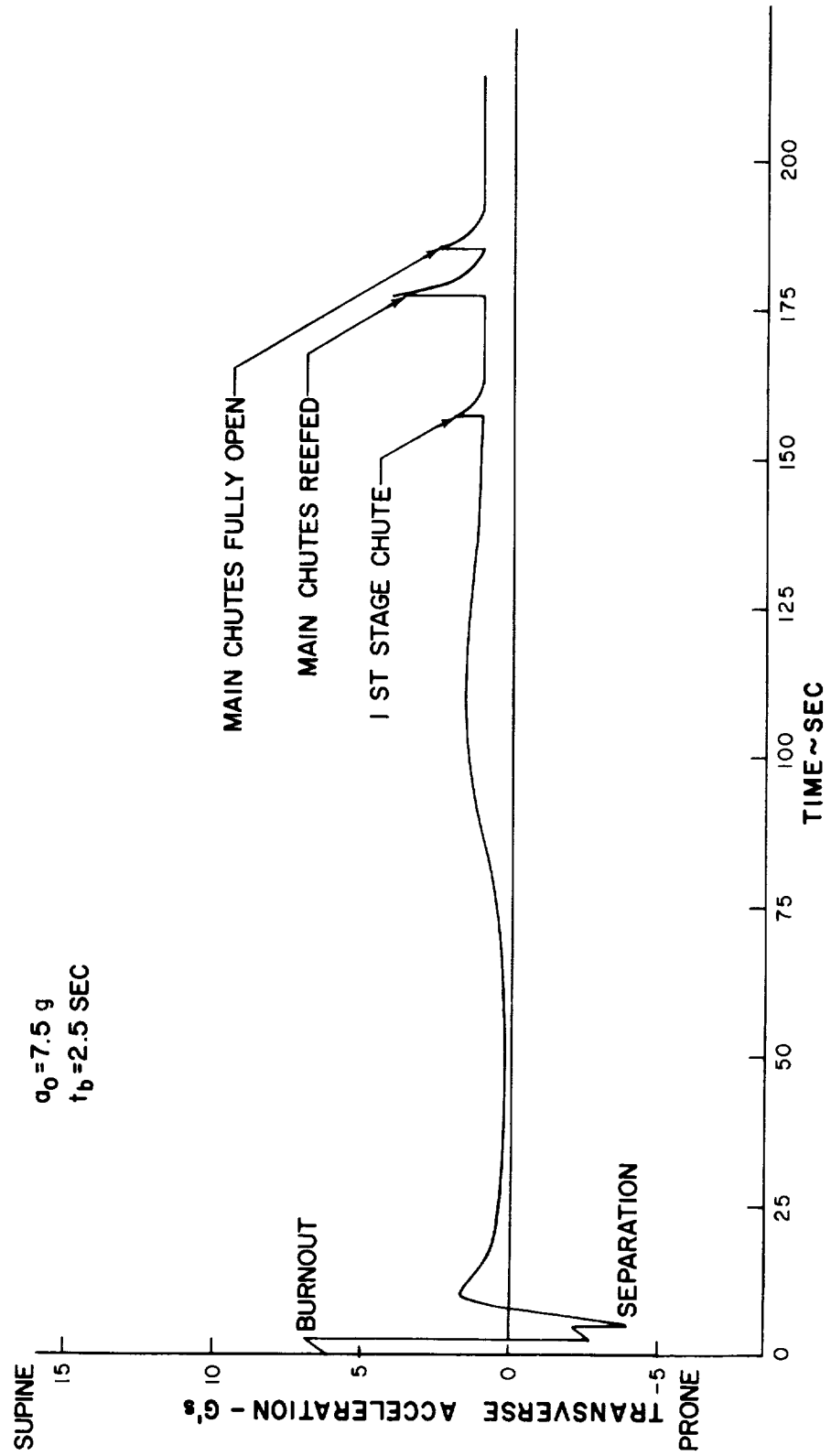


Figure III-4-28. Abort acceleration profile at Saturn S-II ignition
D-2 Configuration

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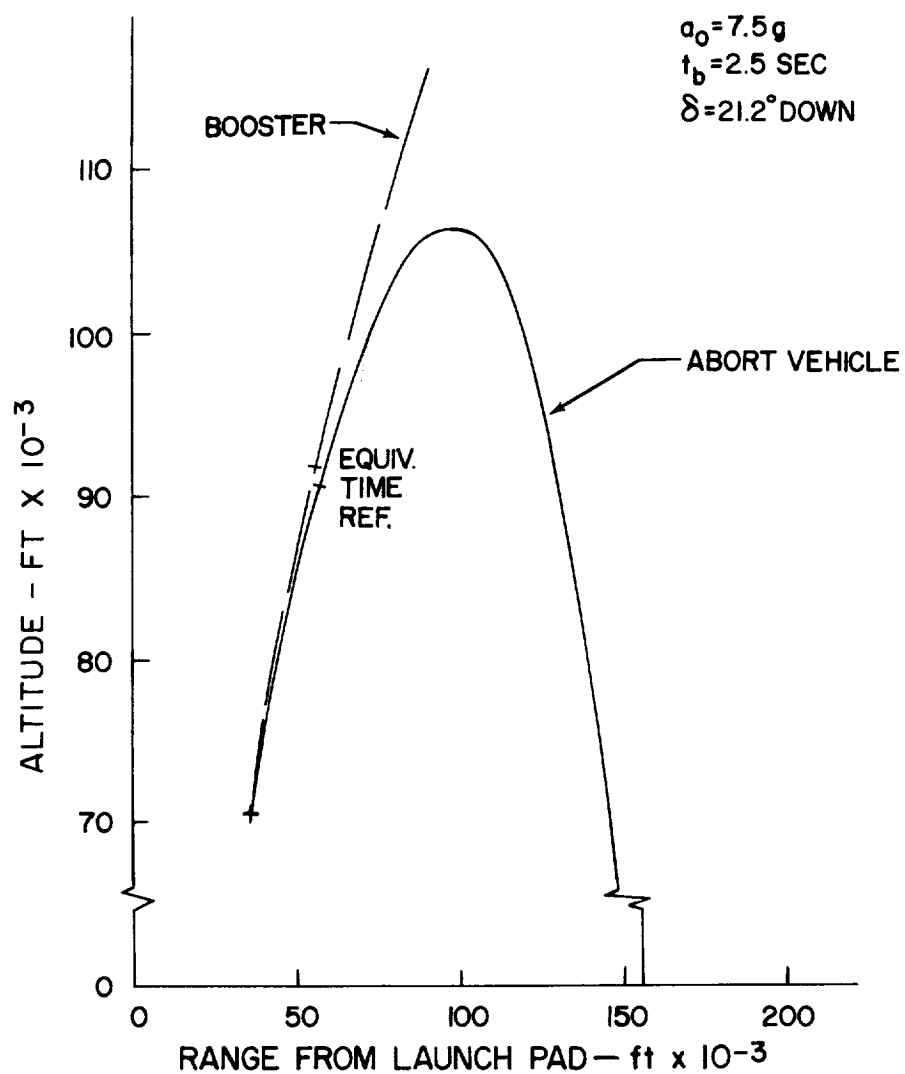


Figure III-4-29. Abort trajectory at Saturn S-II ignition
D-2 Configuration

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| | |
|---------------------------------------|----------------------|
| Initial acceleration: | 6.2 g |
| Burnout acceleration: | 6.8 g |
| Abort rocket thrust inclination: | 21.2 deg. |
| Rocket burning time: | 2.5 sec |
| Number of abort rockets: | 4 |
| Total initial abort rocket thrust: | 77800 lb. |
| Apogee altitude: | 106,448 ft. |
| Velocity at burnout: | 2810 ft/sec. |
| Chute deployment initiation altitude: | 25,000 ft. |
| Range at 25,000 ft: | 157,029 ft (no wind) |
| Acceleration increment: | 9.5 g |

4.1.5 Stage S-II Burnout

The maximum acceleration of the Saturn vehicle occurs just prior to stage S-II burnout. The APOLLO abort vehicle, utilizing four of the original eight abort rockets, has sufficient available acceleration to meet the abort requirements at this time.

4.1.5.1 CRITERIA

At this phase neither the booster explosion or aerodynamic drag criteria affect the abort sequence as the vehicle is travelling at hypersonic velocity with effectively zero dynamic pressure. The conditions at stage S-II burnout, are as follows:

| | |
|----------------------|---------------------------|
| Altitude | 393,720 ft |
| Velocity | 19970.0 ft/sec. |
| Dynamic pressure | 0 |
| Path angle | 85.0 deg. (from vertical) |
| Time from launch | 284.9 sec |
| Range | 1,397,706 ft. |
| Booster acceleration | 5.18 g |
| Weight | 155,000 lbs. |

a. Recovery Systems Operation - The normal re-entry recovery system operating sequence is utilized for abort at, or just prior to, Saturn S-II burnout. The forward and aft space vehicle structure is separated 4.5 seconds after abort and the re-entry vehicle follows a ballistic flight path until parachute deployment at 25,000 feet.

b. Acceleration Profile - The acceleration profile for abort at this phase is shown in fig 4-30. With the vehicle essentially out of the atmosphere, the ballistic re-entry path utilized gives a maximum acceleration of 13.4 g at 122,584 ft. This is not considered beyond the capability of the crew for an abort situation.

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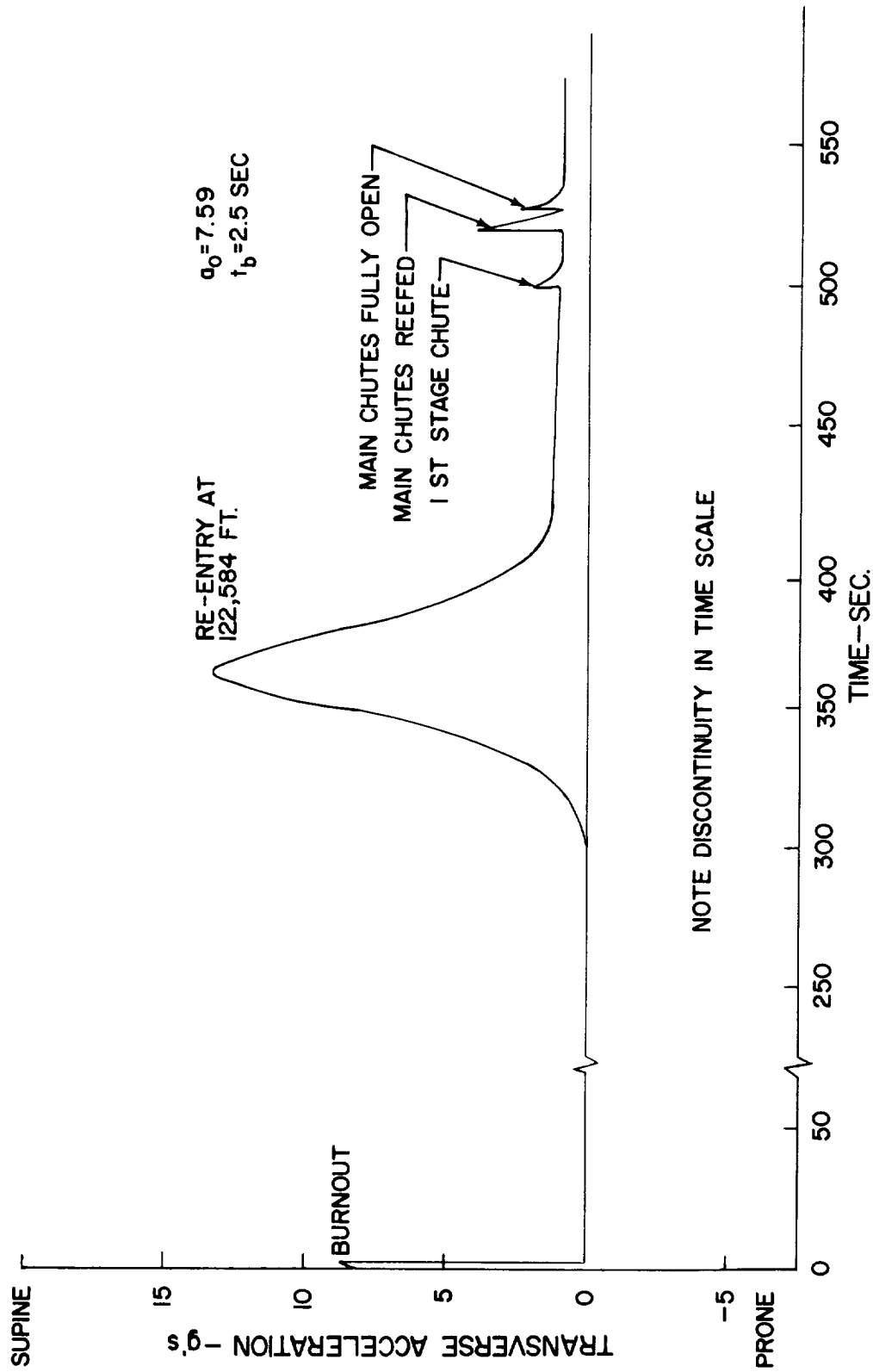


Figure III-4-30. Abort acceleration profile at Saturn S-II burnout
D-2 Configuration

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4.1.5.2 SELECTED SYSTEM PERFORMANCE

The trajectory for abort at stage S-II burnout is presented in Figure III-4-31 with characteristics as follows:

| | |
|---------------------------------------|------------------------|
| Initial acceleration: | 8.1 g |
| Burnout acceleration: | 8.7 g |
| Abort rocket thrust inclination: | 21.2 deg |
| Rocket burning time: | 2.5 sec |
| Number of abort rockets: | 4 |
| Total initial abort rocket thrust: | 77,800 lb. |
| Apogee altitude: | 478,069 ft |
| Velocity at burnout: | 18662 ft |
| Re-entry acceleration: | 13.4 max. |
| Chute deployment initiation altitude: | 25,000 ft |
| Range at 25,000 ft: | 8,105,039 ft (no wind) |
| Acceleration increment: | 8.7 g |

4.1.6 Saturn Stage S-IV Sub-Orbital Phases

After burnout and separation of the Saturn stage S-II vehicle, the abort interface of the APOLLO vehicle is shifted to the APOLLO/S-IV booster interface. Abort during this phase will utilize the two remaining abort rockets for separation from the booster, then utilize the on-board propulsion capabilities to affect an impact in the vicinity of Ascension Island.

Just after S-II separation, the initial point of this phase, the velocity increment required to impact in the specified area is approximately 4800 ft/sec through the longitudinal axis of the vehicle. This is well within the capabilities of the on-board propulsion system, both in level of thrust and total impulse available. As the booster velocity increases the abort requirement, at any given time, decreases until at orbital velocity a velocity increment of approximately -500 ft/sec is required, along the vehicle longitudinal axis, to provide a re-entry trajectory impacting in the Ascension Island area.

The exact impact point for any trajectory is, of course, dependent upon the launch angle and type of re-entry. For this study a launch angle of 108° was used with an equilibrium glide re-entry at $L/D = 0.6$ starting at 400,000 ft. Insertion data were obtained from Reference 7, re-entry data from Reference 8. Figure III-4-32 shows the impact points with the application of various velocity vector increments at Saturn S-II burnout.

Ref 7 J. T. Markley, Saturn C-1 and C-2 Booster System. NASA Project APOLLO Working Paper No. 1002, Nov. 9, 1960.

Ref 8 B. A. Galman, Some Fundamental Considerations for Lifting Vehicles in Return from Satellite Orbit. GE-MSVD TIS Report no. R59SD355, May 7, 1959.

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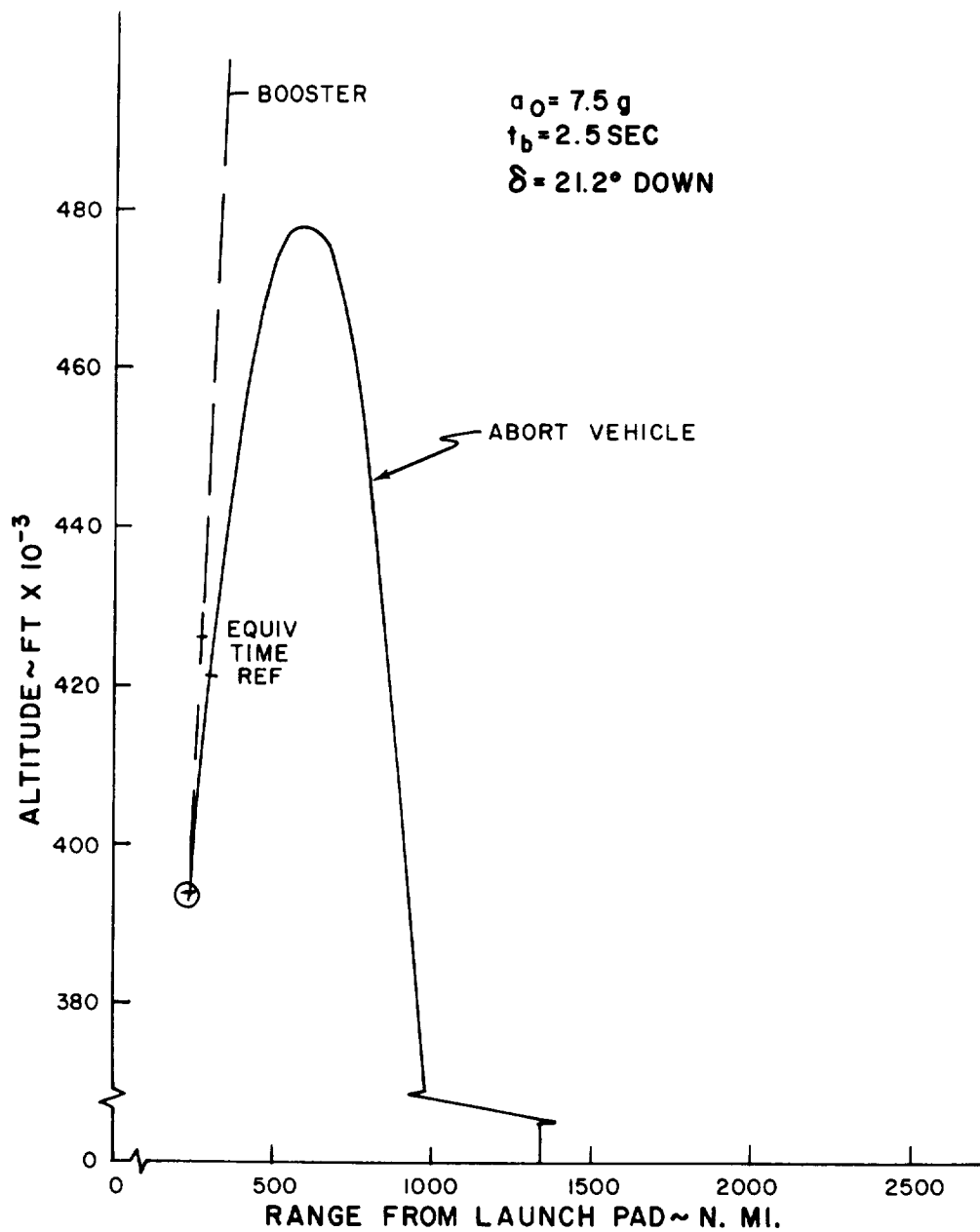


Figure III-4-31. Abort trajectory at Saturn S-II burnout
D-2 Configuration

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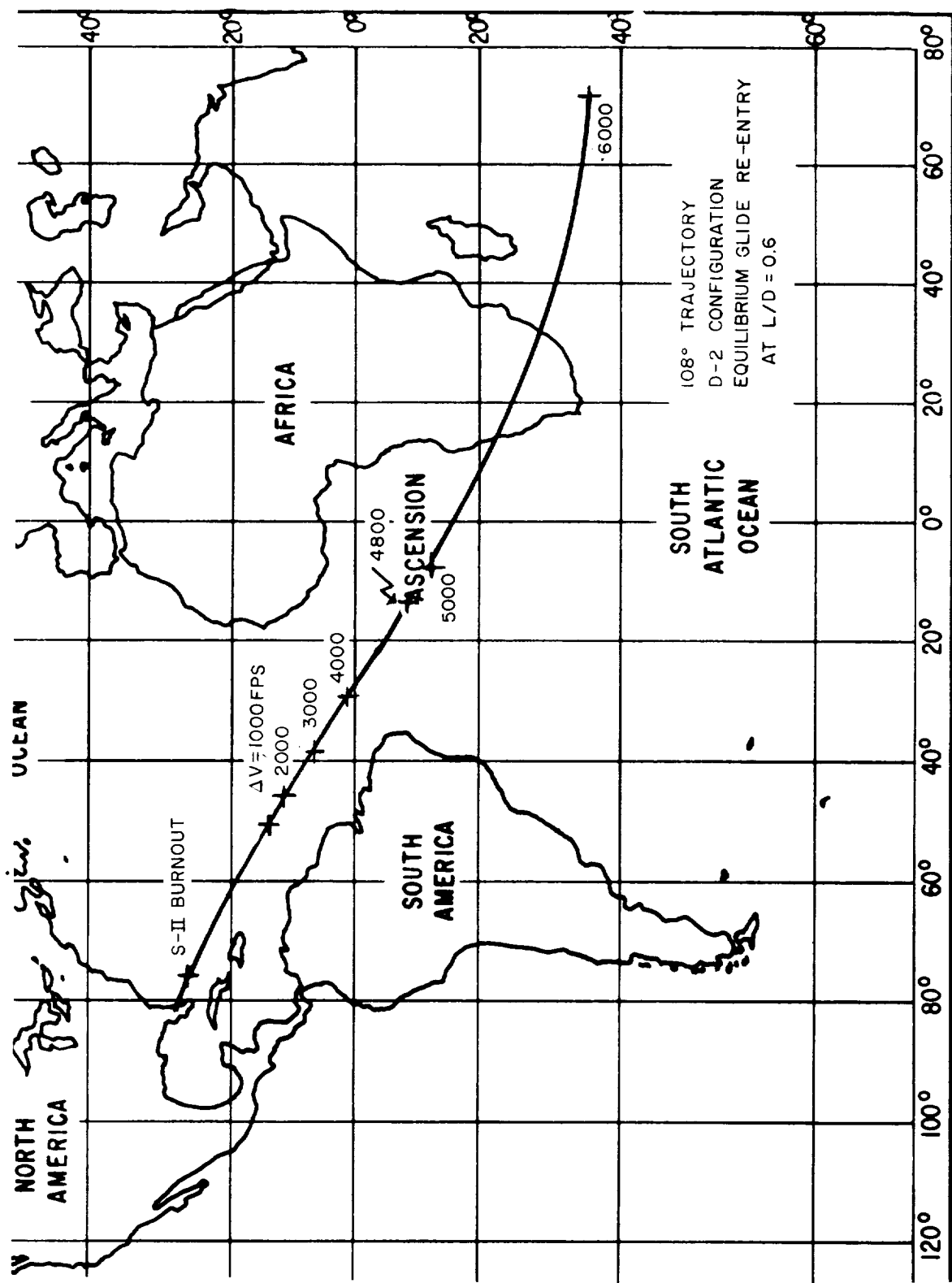


Figure III-4-32. Location of impact points for various additional velocity vector increments at Saturn stage S-II burnout
D-2 Configuration

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4.1.7 Saturn Stage S-IV Super-Orbital Phase

During a major portion of this phase it will be possible, in the event of an abort requirement, to initially change the trajectory to earth orbital and then, at any desired time, de-orbit and land at a selected site. This type of abort can only be utilized until the pre-abort vehicle velocity has reached approximately 32,000 ft/sec. This latter velocity requires the use of on-board propulsion to attain a retro increment of 6500 ft/sec, leaving approximately a 1000 ft/sec velocity increment to de-orbit.

Beyond 32,000 ft/sec velocity, the abort trajectory will be a direct re-entry type emergency return. This type maneuver is described in detail in Chapter 1 of Volume III.

It should be noted that the method of application of the abort velocity vector has a marked influence on the apogee altitude and time for return of the abort vehicle. Data recently made available (reference 9) indicates that the method utilized for this study, direct retrograde along the velocity vector, will result in a desirable time for return of less than 5 hours at velocities up to that of injection. However, utilization of this method of abort propulsion orientation is also shown to result in re-entry conditions which exceed the skip boundary limitation for abort at velocities exceeding approximately 28,000 ft/sec and is relatively insensitive to the magnitude of the abort propulsion increment. It can therefore be seen that the optimum velocity vector orientation for this phase must be determined for the specific trajectory under consideration and will, for most cases, result in a compromise set by the limitations of the re-entry corridor and the minimum return time desired.

4.2 EMERGENCY RETURN

After insertion into the cislunar trajectory, the emergency return modes detailed in Chapter 1 of Volume III will be utilized. The abort computer will determine the desired velocity vector increment required to change to an immediate return (abort), emergency return (selected landing site), or secondary mission trajectory.

4.3 ZONING OF ABORT

The various abort trajectories developed in the preceeding pages of this section can be categorized to form a series of abort zones corresponding to major events or sequences in the flight trajectory. Landing site requirements are minimized by selecting a single site or area for each segment of the boost trajectory as shown on Figure III-4-33.

Ref 3 Eggleston, J. M. and McGowan, W. A., A Preliminary Study of Some Abort Trajectories Initiated During Launch of a Lunar Mission Vehicle. NASA TM X-530, Feb. 21, 1961.

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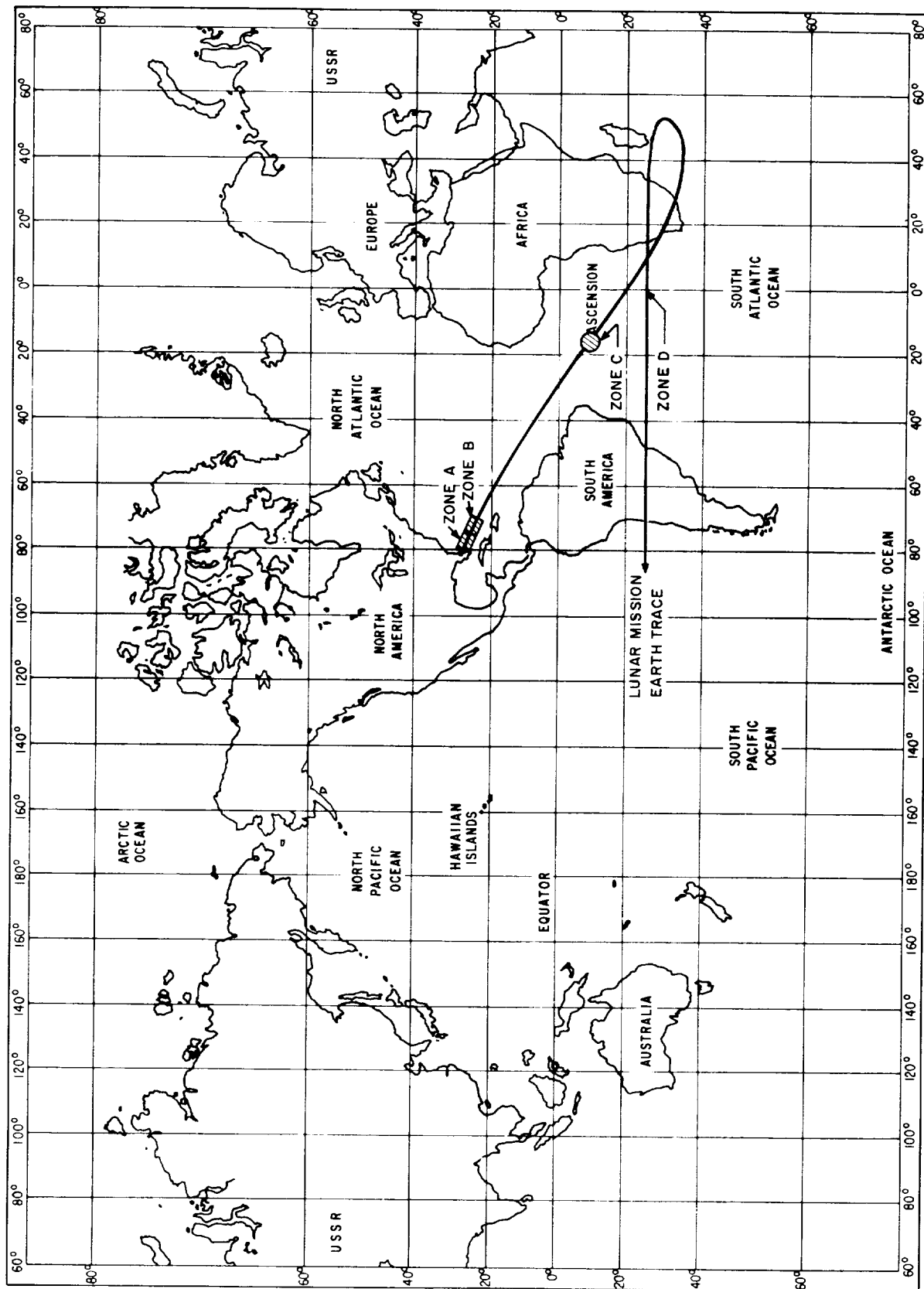


Figure III-4-33. Boost abort zones

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4.3.1 Zone A

From launch pad to burnout of the Saturn S-I booster, the impact area of an aborted D-2 configuration vehicle will be in an area along the earth trace of the boost trajectory with a maximum range at impact of approximately 33 nmi. Because of the relatively short distances involved, rapid pickup and recovery of the crew of the abort vehicle can be made in this area with a minimum of surface vessels (probably only one stationed at the extreme end of this area) plus land based aircraft and shallow draft vessels.

4.3.2 Zone B

During stage S-II boost, the tilt angle of the trajectory increases quite rapidly culminating, for the trajectory utilized (Ref 7), in an angle of 85 degrees from the vertical at S-II burnout. For an abort occurring during stage S-II boost this, together with the increasing velocity of the boosted vehicle, causes the range to impact to increase quite rapidly, reaching a maximum of 1333 nmi for a ballistic re-entry at stage S-II burnout. This area categorized as Zone B is, fortunately, well covered with tracking media utilized for present day IRBM and ICMB testing.

The number of surface vessels and aircraft required in this area should be determined on the basis of past experience in projects Mercury and Discoverer.

4.3.3 Zone C

The third zone of abort is a recovery area near Ascension Island that will be utilized during the boost phase starting at S-IV ignition and terminating when orbital velocity is reached. During this phase it is possible, utilizing the Apollo on-board propulsion, to modify the ballistic portion of the flight trajectory sufficiently to effect a landing in the designated area after an equilibrium glide at $L/D = 0.6$ (see Section 4.1.6). Therefore, no additional landing sites will be required between the end of Zone B and the impact area for Zone C.

4.3.4 Zone D

The tremendous variety of trajectory paths which can be followed for earth orbital, lunar orbital, and circumlunar flights, coupled with variations in insertion angle and the possibility of power-off coasting during the S-IV stage, makes the selection of specific abort landing areas quite difficult, if not impossible, in a study program such as this. It is, however, very probable that, except for abort missions requiring immediate return to earth without regard for landing site, most emergency return trajectories can, utilizing on-board propulsion, return to a selected landing site such as Edwards AFB. This may be

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done directly by making mid-course corrections and re-entering the earth's atmosphere at the proper point for landing, or indirectly by first establishing an earth orbit and then utilizing the on-board propulsion to de-orbit at a selected time and location. Therefore, no specific impact area or zone has been selected for abort or emergency return at superorbital velocities.

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5.0 Special Study Areas

5.1 BOOSTER EXPLOSION - ON LAUNCH PAD

The most critical phase of the APOLLO mission for abort action occurs when the vehicle is on the launch pad. At this time the APOLLO spacecraft is positioned on the 3-stage Saturn booster containing upwards of one million pounds of highly volatile fuel. Although the emergency escape rocket system is designed to the acceleration tolerance of the crew, the altitude requirements of the recovery system, the range requirements for local obstacle avoidance, and the weight of the abort propulsion system, the limiting parameter is the time-distance displacement required of the abort vehicle in the event of a booster explosion.

The data presented at the mid-term review (reference 10) and in the mid-term report (reference 11) were, in a large part, based on booster explosion data contained in WADD Technical Report 60-75, Parts I and II (references 12 and 13). These data analyzed the escape requirements for boosters up to and including the Atlas and, as in this report, did not attempt to define the cause of the explosion but simply assumed that an explosion occurred. An explosion defines a reactive process of high energy releases taking place

Ref 10 Anon., APOLLO Systems Study, Mid-Term Review. General Electric Co., Missile and Space Vehicle Dept., March 3, 1961

Ref 11 Anon., Project APOLLO Data Book. General Electric Co., Missile and Space Vehicle Dept., March 14, 1961.

Ref 12 W. H. Baier and H. M. Pernini, Investigation of Emergency Escape Under Conditions of Extremely High Altitudes and Velocities — Part I: Basic Study Report. WADD Technical Report 60-75, Part I, September 1960.

Ref 13 W. H. Baier and H. M. Pernini, Investigation of Emergency Escape Under Conditions of Extremely High Altitudes and Velocities — Part II: Expanded Scope Report. WADD Technical Report 60-75, Part II, Unissued as of this date.

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in a short period of time, which can be as long as several seconds. However, the only explosions which produce shock waves are high-order detonations. These are reactive processes which take place in periods as short as micro-seconds. Since the booster system under consideration uses types of fuel which could produce a detonation, this study was therefore limited to the investigation of shock wave effects. This is not considered a restrictive parameter since an abort vehicle designed to withstand the shock wave effects of a high order detonation will withstand the effects of any type of explosion.

In order to apply the data to the Saturn C-2 booster it was necessary to determine the percentage of the total fuel which detonates and the equivalent yield of the reacting fuel. Reference 12 indicates that, for vehicles up to the size of the Atlas, approximately one-half of the fuel present in the booster units reacts in an explosion with the remainder usually consumed in subsequent fires initiated by the explosion. This same reference indicates that for boosters of this type one pound of reacting fuel is considered equivalent to one pound of TNT. Thus, the overall yield of a booster explosion is indicated as the TNT equivalent of 50 percent of the total weight of fuel on board. However, because of the large concentration of fuel in the Saturn C-2 vehicle, this value was increased to 75 percent for the data presented in the mid-term review and reports.

Immediately after presentation of this material, Saturn data (reference 14) was released which indicated that the yield of a Saturn C-2 booster explosion was officially estimated as 10 percent equivalent TNT for the propellant of each stage. This reduced the effective yield over that previously utilized by a factor of 7.5. However, during a meeting with Mr. James W. Carter of the Future Projects Office, Marshall Space Flight Center at Huntsville, Alabama on April 13, 1961, we were informed that the 10 percent figure was an erroneous initial estimate. It was then stated that, as of this date, the estimated explosive yield in terms of equivalent pounds of TNT of the Saturn C-2 booster, is 50 percent of the total fuel remaining in all stages at the time of the explosion. This data, which is in direct agreement with that presented in references 12 and 13, has been utilized for this final report.

Ref 14 Anon., Preliminary Saturn General Information. NASA Space Task Group, Langley Field, Va., Feb. 28, 1961.

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The total fuel contained within the Saturn C-2 just prior to launch was obtained from reference 14 and is as follows:

| | |
|-----------------------|------------------------|
| S - I Stage | 650, 000 lbs |
| S - II Stage | 330, 000 lbs |
| S - IV Stage | 100, 000 lbs |
| Total Saturn C-2 Fuel | <u>1, 080, 000 lbs</u> |

Using the previously described criteria for determining the equivalent TNT yield we obtain a reactive equivalent of 275 tons of TNT. For a 1000 lb (1 KT) explosion of TNT, the distance at which the maximum overpressure does not exceed 10 psi is 975 feet. (reference 15).

Utilizing the expression:

$$d = d_o (W)^{1/3} \quad (1)$$

where d_o = reference distance from the explosion source
for a 10 psi overpressure for a 1-kiloton
explosion, ft.

W = equivalent reactive TNT, lbs

$$d = 975 (.275)^{1/3} = 630 \text{ ft.}$$

As the APOLLO abort vehicle is located on the extreme nose of the Saturn booster it has an initial displacement from the c.g. of the explosion when considering the detonation as a point source explosion. Considering the center of gravity of the explosion to occur at the center of gravity of the propellant, it can be seen from Figure III-5-1 that this occurs in the vicinity of 6 feet below the S-I/S-II interface. Introducing a slight factor of conservatism, we have selected this interface as the launch pad booster explosion c.g., or point source. The APOLLO abort vehicle is located 106 feet above the point. Therefore, the vehicle must be moved a distance of

$$d = 630 - 106 = 524 \text{ ft.}$$

from the Saturn C-2 booster in order to assure that the maximum overpressure of 10 psi is not exceeded.

Ref 14 Anon., Preliminary Saturn C-2 Information, NASA Space Task Group, Langley Field, Va., Feb. 28, 1961.

Ref 15 S. Glasstone, The Effects of Nuclear Weapons. United States Atomic Energy Commission, June 1957.

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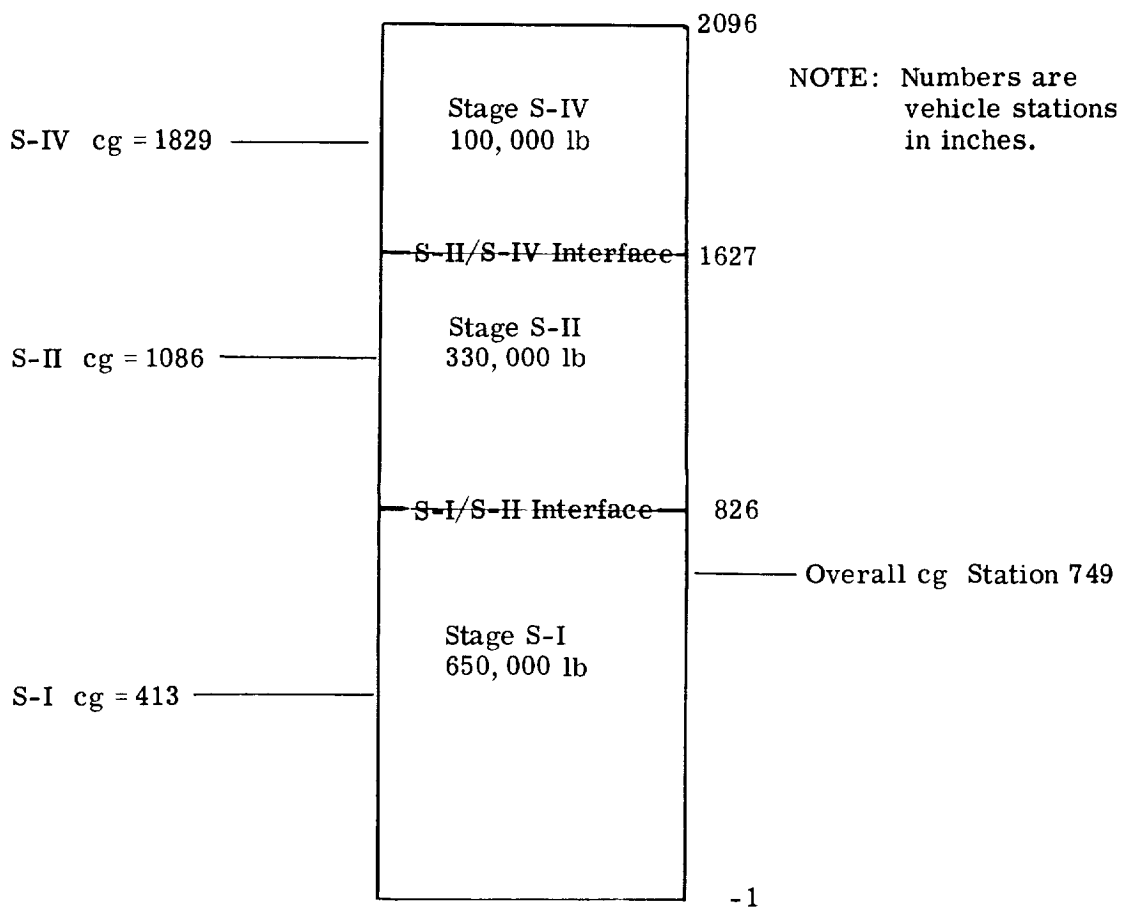


Figure III-5-1. Saturn C-2 fuel distribution on launch pad

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Again referring to reference 15, the time for the 10 psi shock wave to propagate to the distance d_0 for the 1-kiloton explosion is 0.44 seconds. As

$$t = t_0 (W)^{1/3} \quad (2)$$

where t_0 = reference time from the explosion source for a
10 psi overpressure for a 1-kiloton explosion, sec.

W = equivalent reactive TNT, lbs.

$$t = 0.44 (.275)^{1/3} = 0.28 \text{ sec.}$$

It can be readily seen that the requirement for moving the APOLLO abort vehicle a distance of 524 feet in 0.28 seconds would only occur in the event that the booster exploded at the same instant that the APOLLO vehicle was aborted. Not only is this an extremely unlikely occurrence as it assumes zero warning time, but it would impose prohibitive accelerations in excess of 400 g on the crew and vehicle. It is therefore necessary to assume, or select, a sufficient warning time in order that an abort can safely be accomplished without imposing intolerable accelerations upon the crew nor adding considerably to the escape rocket weight requirements. However, it must be remembered that as the minimum time requirement is increased, complexity is added and reliability reduced in the warning system.

A mathematical analysis of this problem is now under consideration at the General Electric Co. Using the methods of calculus of variations and bounding the problem with the various limiting criteria for this type of abort, it is planned to arrive at an optimum warning time for the system. This can then be applied as a requirement for the design of an abort computer for the APOLLO vehicle. Until such analysis is completed, and for the purposes of this report, the available information (references 13 and 16) indicates that the minimum time for pressure build-up in the booster unit tanks prior to an explosion is approximately 2 seconds. If 0.5 second is allowed for sensing, actuation of the abort mechanism, and thrust build-up in the solid propellant abort rockets, the abort action for the minimum warning case occurs 1.5 seconds before the explosion. Using this as a criteria the time allowable for the abort vehicle to move 524 feet from the booster is

$$t = 0.28 + 1.5 = 1.78 \text{ sec.}$$

Ref 16 Address by W. von Braun, Symposium on Aviation Medicine and Space Travel.
November, 1958.

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Under vacuum conditions this requires a longitudinal acceleration of 10.3 g on the abort vehicle. Due to aerodynamic drag this figure will necessarily be increased somewhat.

5.2 BOOSTER EXPLOSION - MAX. q CONDITIONS

Abort during powered flight at the point of maximum dynamic pressure (max q) is another critical case. This is a result of increased drag which considerably reduces the accelerating effect of the abort rockets. Fortunately, the booster has by this time consumed a considerable portion of the S-I stage fuel, thereby reducing the explosive yield. In addition, for reduced atmospheric pressure and temperature, the overpressure at a given distance from an explosion of specified yield generally decreases. Therefore, the radius from the explosion source to the 10 psi maximum overpressure point decreases. Data from reference 17 indicates that at 32,800 feet altitude (max q), the fuel remaining in the booster is approximately 743,700 lbs. Using the same criteria as for the launch pad case we can calculate, for an air blast at 32,800 feet, the distance required in order to limit the maximum overpressure to 10 psi.

$$d = d_o (W)^{1/3} \left(\frac{P_o}{P} \right)^{1/3} \quad (3)$$

where:

P_o = standard pressure at sea level

P = ambient pressure at blast altitude

$$d = 425 (0.186)^{1/3} (3.871)^{1/3} = 381 \text{ feet}$$

As shown in Figure III-5-2, the c. g. of the propellant for the max q abort condition is at vehicle station 870. This is slightly above the S-I/S-II stage interface. Again introducing a slight conservatism, we have assumed that the explosion c. g. for this condition occurs at the S-II tank location, station 1032. This places the APOLLO abort vehicle 89 feet above the point of assumed explosion. The vehicle must therefore move a distance of

$$d = 381 - 89 = 292 \text{ feet}$$

in order to limit the maximum overpressure to 10 psi.

Ref 17 J. T. Markley, Saturn C-1 and C-2 Booster System. NASA Project APOLLO Working Paper No. 1002, November 9, 1960.

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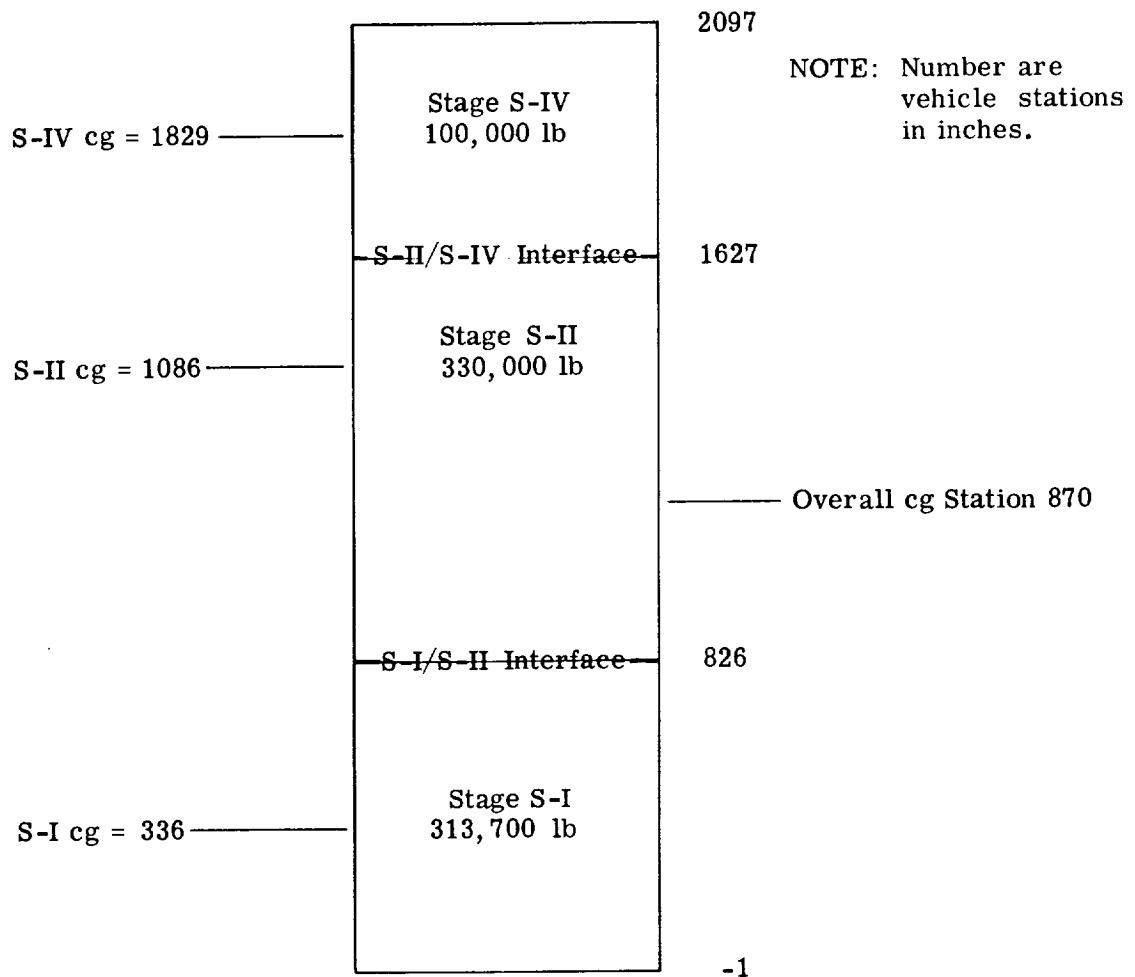


Figure III-5-2. Saturn C-2 fuel distribution max q condition

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The time for the 10 psi shock wave to intercept the vehicle at this point is given by the expression

$$t = t_o (W)^{1/3} \left(\frac{P_o}{P} \right)^{1/3} \left(\frac{T_o}{T} \right)^{1/2} \quad (4)$$

where:

T_o = Standard temperature at sea level

T = Ambient temperature at blast altitude

$$t = 0.30(01.86)^{1/3}(3.871)^{1/3}(1.294)^{1/2}$$

$$t = 0.31 \text{ seconds}$$

Applying the previously assumed warning criteria we get a total minimum abort time under conditions of maximum dynamic pressure of

$$t = 0.31 + 1.5 = 1.81 \text{ seconds}$$

When the booster has attained a high velocity and, in addition, is undergoing an acceleration, the displacement of the ejected abort vehicle with respect to the booster at the instant of detonation is less than its displacement following a launch pad abort. The explosion shock wave propagates at transonic velocities from the point of detonation, which in itself does not partake of the motion of the booster, and which remains stationary in space. For those aborts made when the booster, and hence the abort vehicle, are traveling at velocities above the transonic range, the abort vehicle will outrun the explosion regardless of the displacement at explosion. This is the case for all explosion aborts above approximately 35,000 feet during ascent.

The term overpressure referred to throughout this study is defined as the transient pressure (manifested in the shock wave from the detonation) in excess of the local ambient pressure. The peak overpressure is quoted since the overpressure decays exponentially with time and distance. The duration of the positive pressure phase of the shock wave at a given location defines the duration of the loading on the engulfed structure at that point. The negative pressure phase is not studied since the loading effects are considered negligible as compared to the loading effects of the positive pressure phase. Based on this consideration, the structure is subjected to the maximum possible loading by the shock wave. The pressure-time curve of the shock wave is assumed to have a zero rise time at any given distance, as shown in Figure III-5-3.

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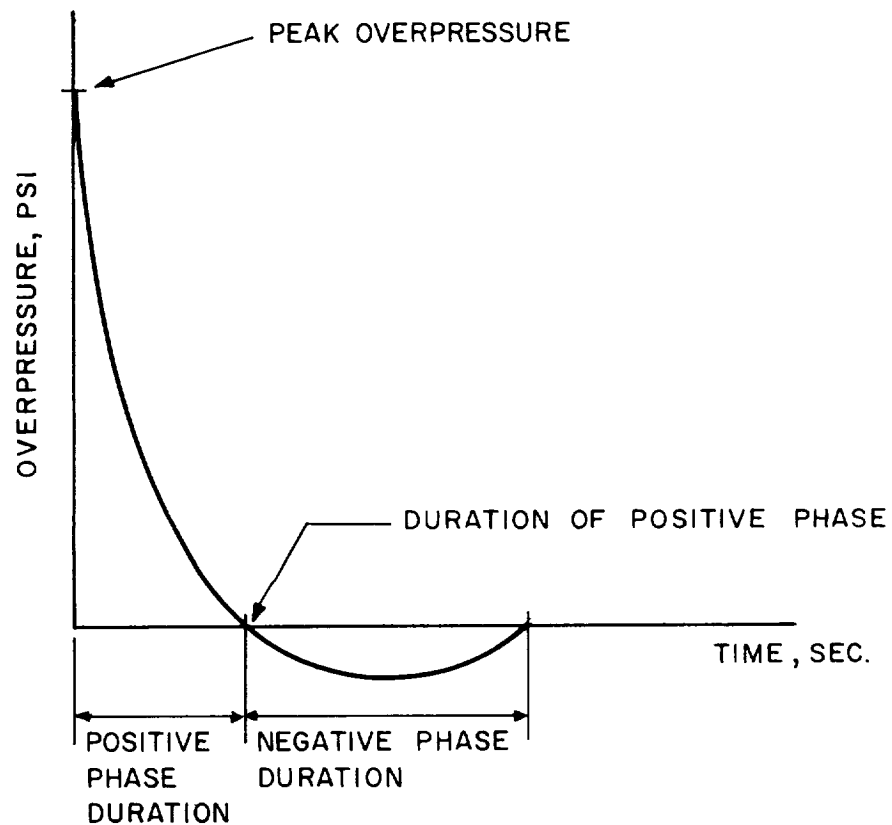


Figure III-5-3. Shock wave pressure-time history at a specified distance from the point of detonation

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In summation, Figure III-5-4 indicates the displacement required to limit the peak overpressure from the exploding booster to a specified limit. Indicated also is the reference curve for a 1-kiloton explosion plus data for the original estimated Saturn yield (75%, references 10 and 11) and the preliminary data received from NASA - STG on March 1, 1961 (10%, reference 14). Similarly, the time required for the shock wave to propagate this distance is shown on Figure III-5-5.

5.3 ABORT STABILITY

5.3.1 Prior to R/V Separation

The re-entry vehicle remains within the aborted space vehicle for a total period of 4.5 seconds (2.0 seconds after abort rocket burnout). An exception is the case of abort beyond Saturn stage S-II burnout, where separation does not occur until just prior to atmospheric re-entry at approximately 400,000 feet. However, for this case, the dynamic pressure at abort is zero and aerodynamic stability need not be considered. The external geometry of the aborted space vehicle is shown in Figure III-5-6.

The center of pressure variation with Mach number for the space vehicle is shown in Figure III-5-7. This stability prediction was obtained from the vast amount of stability data available on flared bodies considered in ICBM and IRBM studies. References 18 to 20 are typical among the reports used in assessing this static stability. The vehicle is observed to be statically stable over its entire flight speed range.

Since the spacecraft is the same vehicle that will be aborted off the launch pad and in flight, trajectory calculations of this vehicle necessitate a knowledge of the zero-lift drag coefficients at $\alpha = 0^\circ$ and variations in angle of attack.

The drag coefficient vs Mach number for the launch abort spacecraft at zero-lift ($\alpha = 0^\circ$), $\alpha = 5^\circ$ and $\alpha = 10^\circ$ are presented in Figure III-5-8. These data were

Ref 18 Laurenson D. I., Summary of Normal Force and Center of Pressure Data on a Variety of Blunt Bodies. GE-MSVD ADM 1:12 Nov., 14, 1958.

Ref 19 Kirk, D. B., and Chapman G. T., The Effectiveness of Conical Flares on Bodies with Conical Noses. NASA TM X-30, Sept. 1959.

Ref 20 Wakefield, R. M., Knichtel, E. D., and Treon, S. L., Transonic Static Aerodynamic Characteristics of a Blunt Cone-Cylinder with Flared Afterbodies of Various Angles and Base Areas. NASA TM X-106, Dec. 1959.

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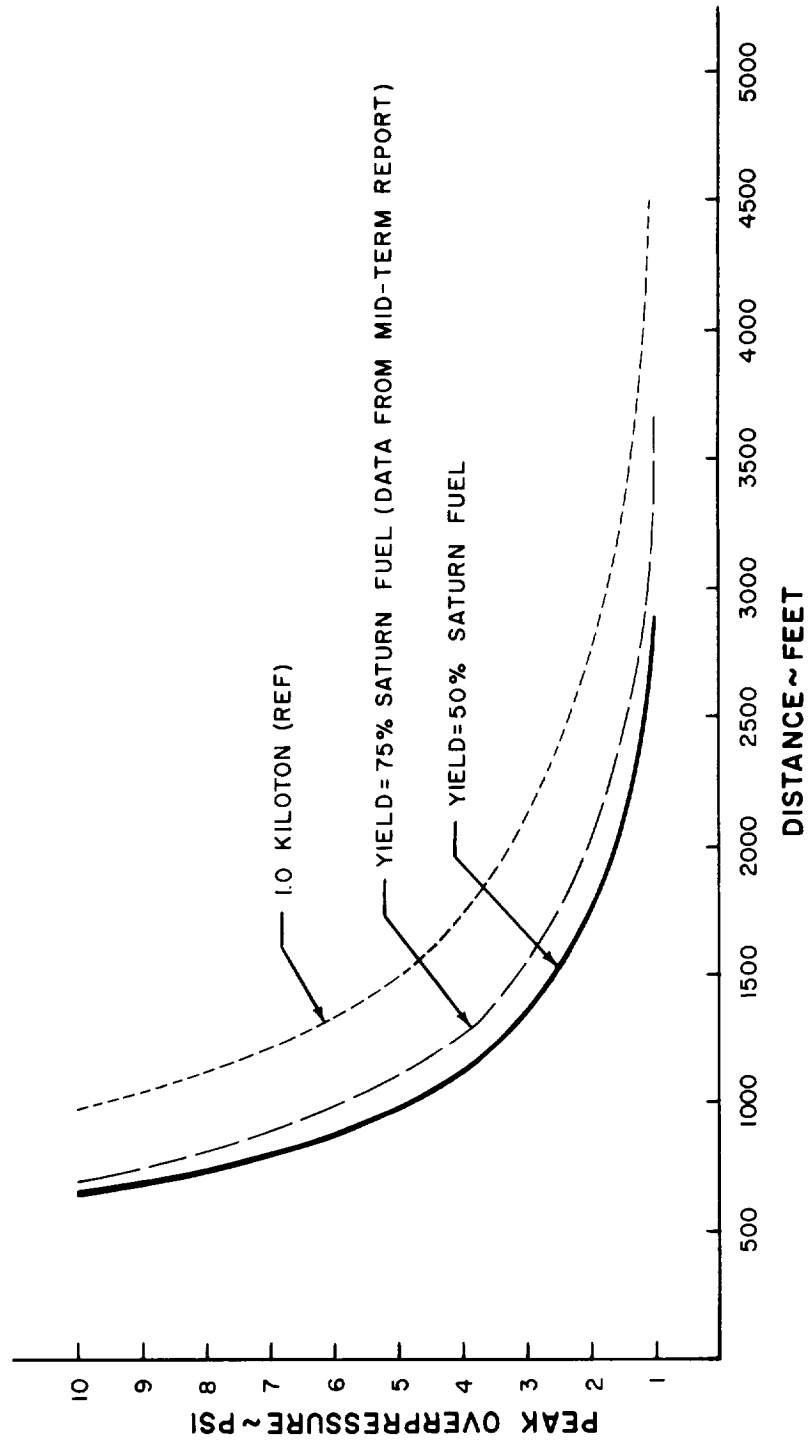


Figure III-5-4. Peak overpressure vs. distance — Launch pad explosion

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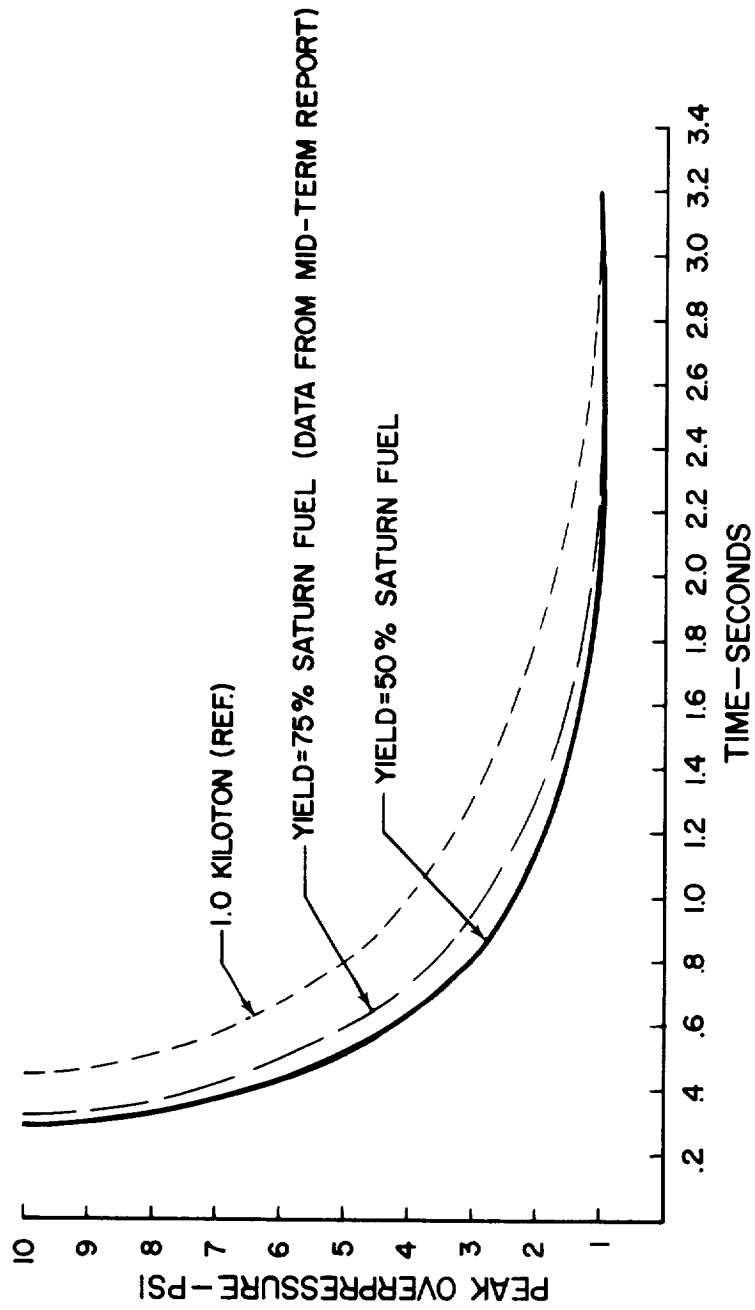
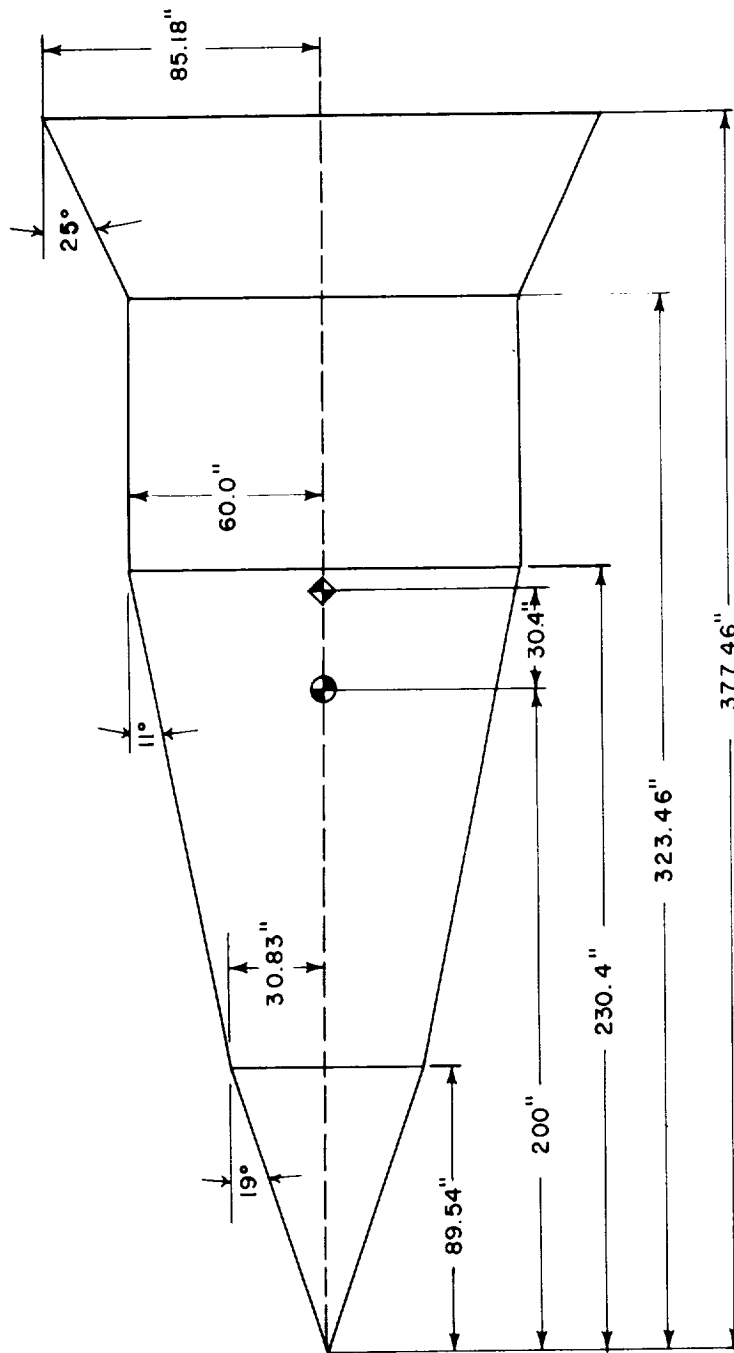


Figure III-5-5. Peak overpressure vs. time —Launch pad explosion

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⊕ X_{CG} WITHOUT ESCAPE ROCKETS
⊕ X WITH ESCAPE ROCKETS

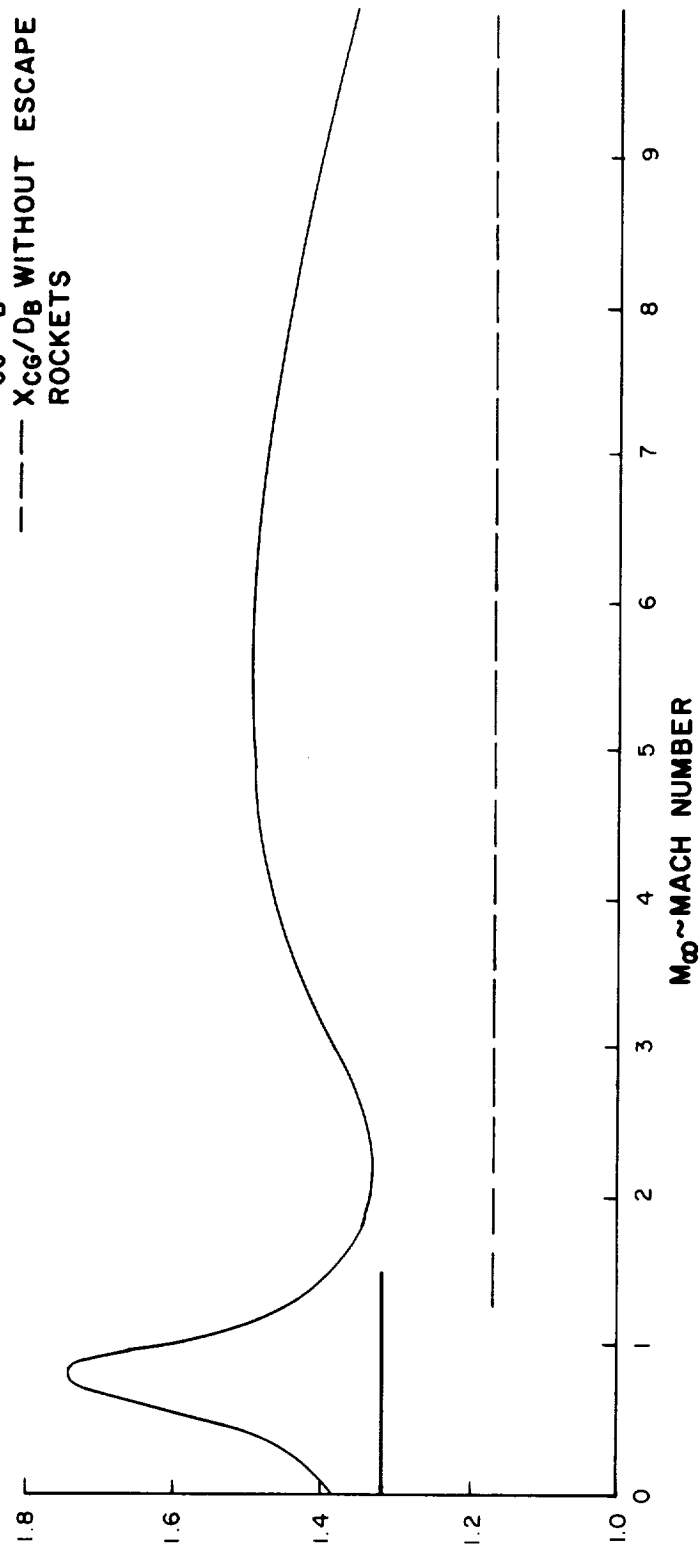


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Figure III-5-6. Launch abort vehicle

CENTER OF PRESSURE~BASE CALIBERS~

$\frac{X_{CP}}{D_B}$ (FROM NOSE)



$\alpha=0^\circ$
 $D_B=14.196$ FT.

Figure III-5-7. Launch abort vehicle.
 Center of pressure location vs. Mach number

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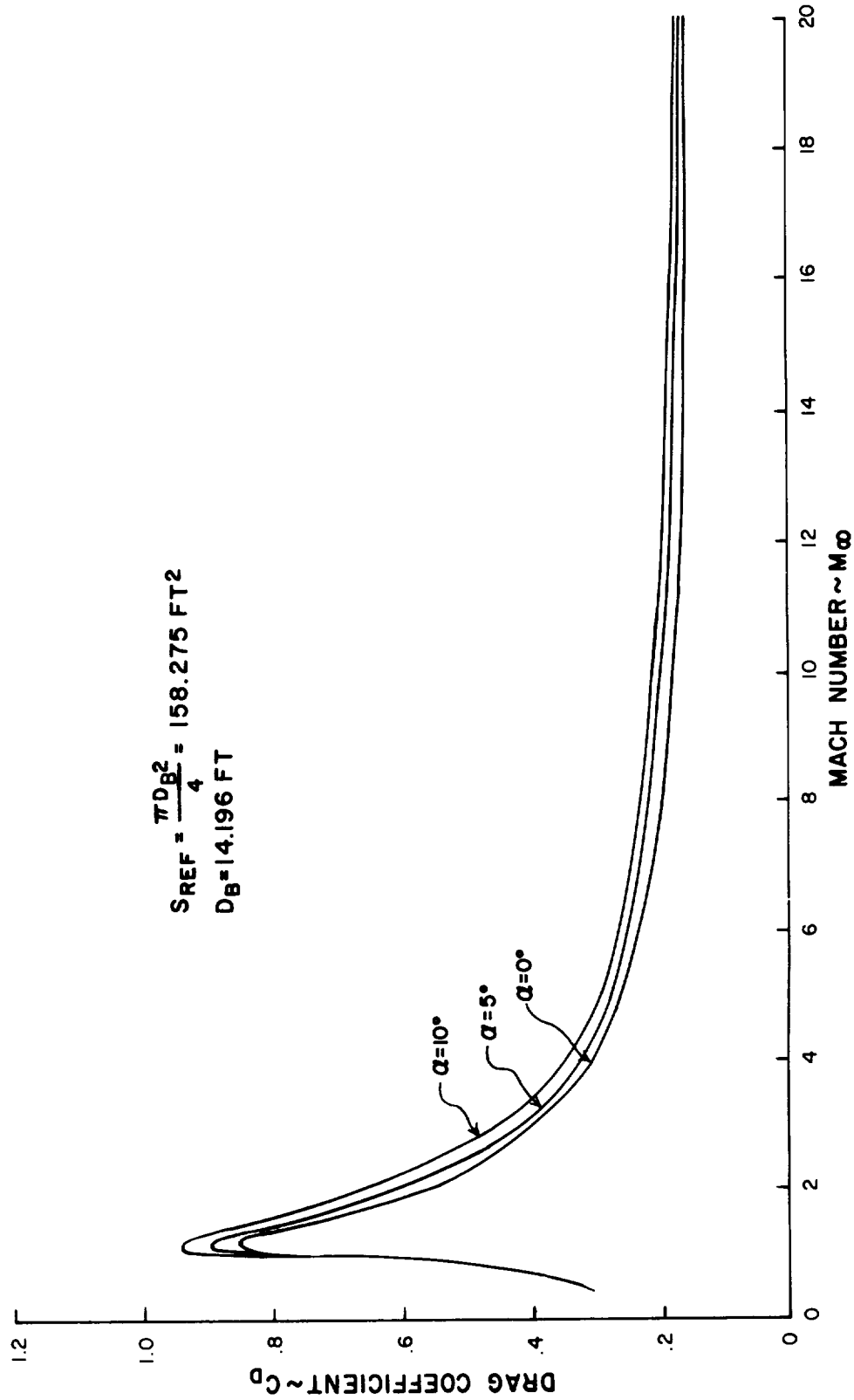


Figure III-5-8. Spacecraft launch abort configuration.
Variation of drag coefficient with Mach number

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obtained using the semi-empirical method prescribed in references 21 and 22 for computing the zero-lift drag coefficient for flared bodies of revolution and then adding the incremental drag due to angle-of-attack effects.

5.3.2 After R/V Separation

The aerodynamic performance of the D-2 and R-3 configuration re-entry vehicles is presented in detail in Volume VI.

5.4 ABORT PROPULSION SYSTEM

A detailed description of the APOLLO propulsion system, including abort propulsion, is contained in Volume IV. Excerpts from this Volume, relative to the abort propulsion system, are presented in the following paragraphs. It should be noted that the subcontractors contributions to Volume IV were based on abort propulsion requirements generated early in the program for the B-2 vehicle. These have subsequently been revised as new Saturn data became available and the D-2 configuration selected. A comparison is as follows:

| | <u>Propulsion Study (Early Reqm'ts)</u> | <u>Final Abort Reqm'ts</u> |
|---------------------------------------|---|--------------------------------|
| Abort weight (exclusive of rockets) | 7000 lbs | 7280 lbs. |
| Number of abort rockets | 8 | 8 |
| Initial abort acceleration | 20 g | 15 g |
| Burning time | 2.0 sec | 2.5 sec |
| Abort rocket mounting angle | 25 ⁰ | 30 ⁰ |
| Net thrust vector through abort c. g. | 15 ⁰ | 20 ⁰ |

5.4.1 Abort Rockets

Early in the study program it was determined that the abort requirements during boost could most satisfactorily be met by solid motors, as typified by their high thrust, short

Ref 21 Laurensen, D. I., The Effect of Mach Number on Aerodynamic Drag - A Compilation of Experimental Data. GE-MSVD ADM 1:14, Sept 8, 1959.

Ref 22 Laurensen, D. I., A Semi-Empirical Method for Preliminary Estimation of the Drag of Axisymmetric Ballistic Re-Entry Vehicles. GE-MSVD ADM 1:24, July 7, 1960.

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duration, short reaction time, and high reliability aspects. It also became obvious that the thrust required of the solid motors will be greatest for launch pad, lift-off and first-stage boost aborts as the thrust requirements continually decrease during the second and third-stage burning. The weight of the launch pad abort propulsion is quite significant, and considerable vehicle weight saving can be achieved by discarding the excessive units during second and third-stage burning. It therefore appears prudent to use multiple abort units. This further enhances the chances of safe abort, for should one unit fail to ignite, the remaining units ensure a reasonable chance of survival. This is true especially in regions of the trajectory where the solid rocket capability is in excess of that which is required. The abort rocket motors are mounted on the vehicle as shown on Figure III-5-9. Four of the eight motors will be jettisoned at first-stage burn out, two at second-stage burn-out and the remaining two at burnout of the third stage of the Saturn booster. Weight penalties which must be charged against the APOLLO vehicle are two percent of the weight jettisoned at first stage burnout and twelve percent of the weight jettisoned at burnout of the second stage of the Saturn booster. All abort propulsion weight carried to third-stage burnout must be charged to APOLLO vehicle weight. Thus, if the jettisoned unit weight is w_j , the abort weight penalty is $(0.02)(4)w_j + (0.12)(2)w_j + 2w_j = 2.32w_j$. The launch pad abort motor weight is $8w_j$ plus the weight of attachments which remain on the vehicle of approximately 32 lbs. Since this weight is small compared with the jettisoned weight, the weight penalty charged against the APOLLO vehicle weight is $2.32/8 = 0.29$ times the launch pad weight. Thus, each pound charged to the vehicle weight represents roughly three pounds of abort rocket weight.

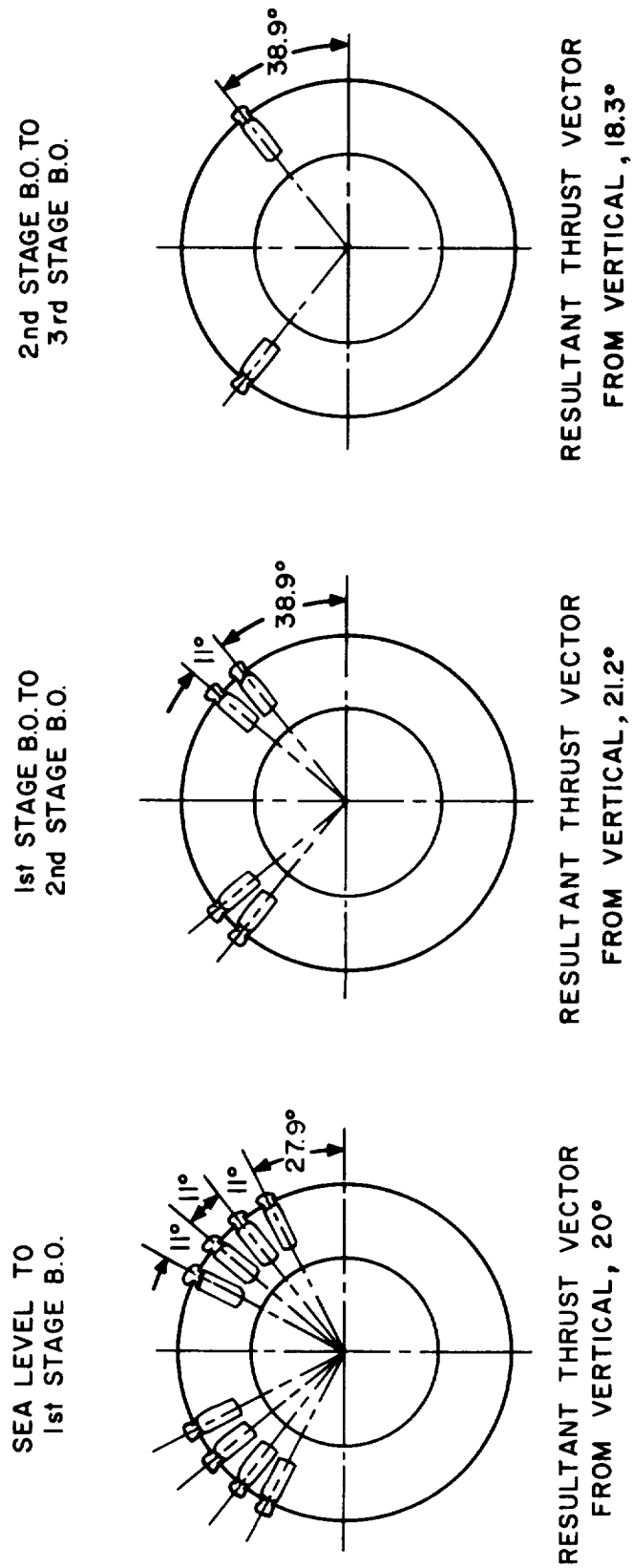
One aspect of the abort propulsion system is its independence of total vehicle weight. The abort sequence which uses the solid motors involves aborting only the re-entry vehicle and mission module while the main on-board APOLLO propulsion remains with the third-stage of the Saturn booster. Consequently, weight of the solid abort rockets is only approximately half that necessary to abort the complete APOLLO vehicle. The total abort motor weight is therefore a function only of re-entry vehicle and mission module weight and is independent of the weight of the main on-board propulsion system.

5.4.2 Large Separation Rockets

Four large separation rockets are utilized to separate the forward space vehicle structure from the re-entry vehicle. This capability is needed only during the high drag regions and it is advisable to jettison these units as soon as they are no longer required.

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Figure III-5-9. Mounting location of abort motors

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This should occur no later than third stage ignition, in which case the penalty charged to the APOLLO vehicle weight will be twelve per cent of the weight which is jettisoned. The use of four motors is favored because of the higher total thrust capability if one unit should fail. The thrust requirement is tailored to the maximum dynamic pressure abort condition and decreases on either side of this point.

5.4.3 Small Separation Rockets

Eight rockets are provided to separate the re-entry vehicle from the on-board propulsion package and mission module prior to re-entry into the earth's atmosphere. These are also utilized for the same purpose in the abort sequence. Four of these are mounted with their nozzles facing forward and are used to separate the on-board propulsion and aft space vehicle structure. The remaining four have aft facing nozzles and separate the mission module and forward space vehicle structure from the re-entry vehicle.

5.4.4 R-3 Configuration Abort Rockets

Solid propellant rocket motors are used on the APOLLO R-3 vehicle for abort only. Six motors are utilized, with four jettisoned at first stage burnout and the remaining two jettisoned at third stage burnout. The penalty factors established earlier show that this jettison sequence has a penalty of $(0.2) (4) w_j + 2 w_j = 2.08 w_j$ where w_j is the jettisoned unit weight. The launch pad abort motor weight is therefore $6 w_j$ plus 24 pounds of attachments which remain on the vehicle. Assuming this attachment weight is small in comparison with the jettisoned weight, the weight penalty charged against the R-3 vehicle is $2.08/6 = 0.39$ times the jettisoned weight. Consequently, only one pound of penalty is incurred for each 2.5 pounds of abort motor weight.

5.5 RECOVERY SYSTEM

5.5.1 Abort Sensing for Recovery Sequence

A multipurpose programmer will be provided to initiate the various functions of recovery such as cover ejection, parachute deployment, marker device actuation, etc. This unit will supply electrical commands at the proper timing intervals for abort as well as normal re-entry conditions.

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A major problem which exists in the design of such a control system is establishing a reference point for initiating a given sequence which will assure successful recovery under all possible flight conditions. This reference point need not be precise with relation to a given set of aerodynamic parameters, but must be within limits which will:

Assure sufficient altitude (or time) for the recovery sequence to be completed prior to impact.

Assure parachute deployment below excessive dynamic pressures or velocities which will afford maximum chute opening reliability and structural load integrity.

Examining a typical terminal trajectory from 100,000 feet altitude to impact for the D-2 configuration (Figure III-5-10) it is apparent that a rather broad operational band exists for a typical post re-entry flight from orbit. Dynamic pressures are relatively low permitting the use of a conventional parachute system and subsonic parachute deployment velocities exist from 73,000 feet altitude to impact.

This, however, is not the case for abort missions. Consequently sensing devices in the control system must be provided which will define the point of recovery equipment initiation within a much narrower operational band for a particular abort case.

Assuming 25,000 feet as an optimum low altitude for recovery sequence initiation, a pressure sensing device will be provided to fix this as the low-altitude limit for the major portion of the flight conditions. This will assure sufficient altitude for completion of the recovery events in all cases except for abort missions either off the launch pad or during the early portion of first-stage powered flight. In every case, however, all recovery event control circuits should be maintained in an unarmed condition until the re-entry vehicle has separated from external structure.

Low altitude ballistic flights resulting from abort conditions during the early portion of boost flight will require a timing interval referenced from re-entry vehicle separation. In this case the altitude sensing device is maintained in an unarmed condition until such time as the re-entry vehicle reaches apogee or is beyond this point on the ballistic flight path. For the launch pad abort case, a timer only would be required to initiate the recovery sequence. Figure III-5-11, showing a typical sequence, indicates that the time interval would be 12.5 seconds from abort rocket ignition to apogee and the first event of the recovery sequence.

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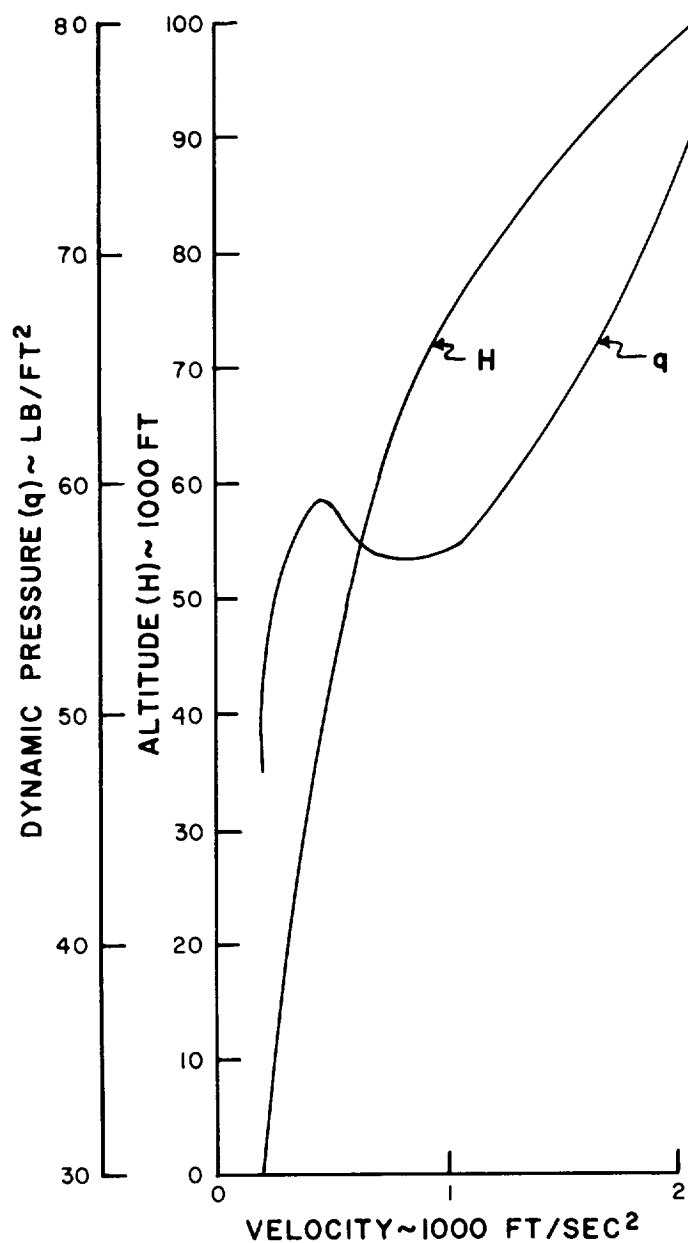


Figure III-5-10. D-2 terminal trajectory

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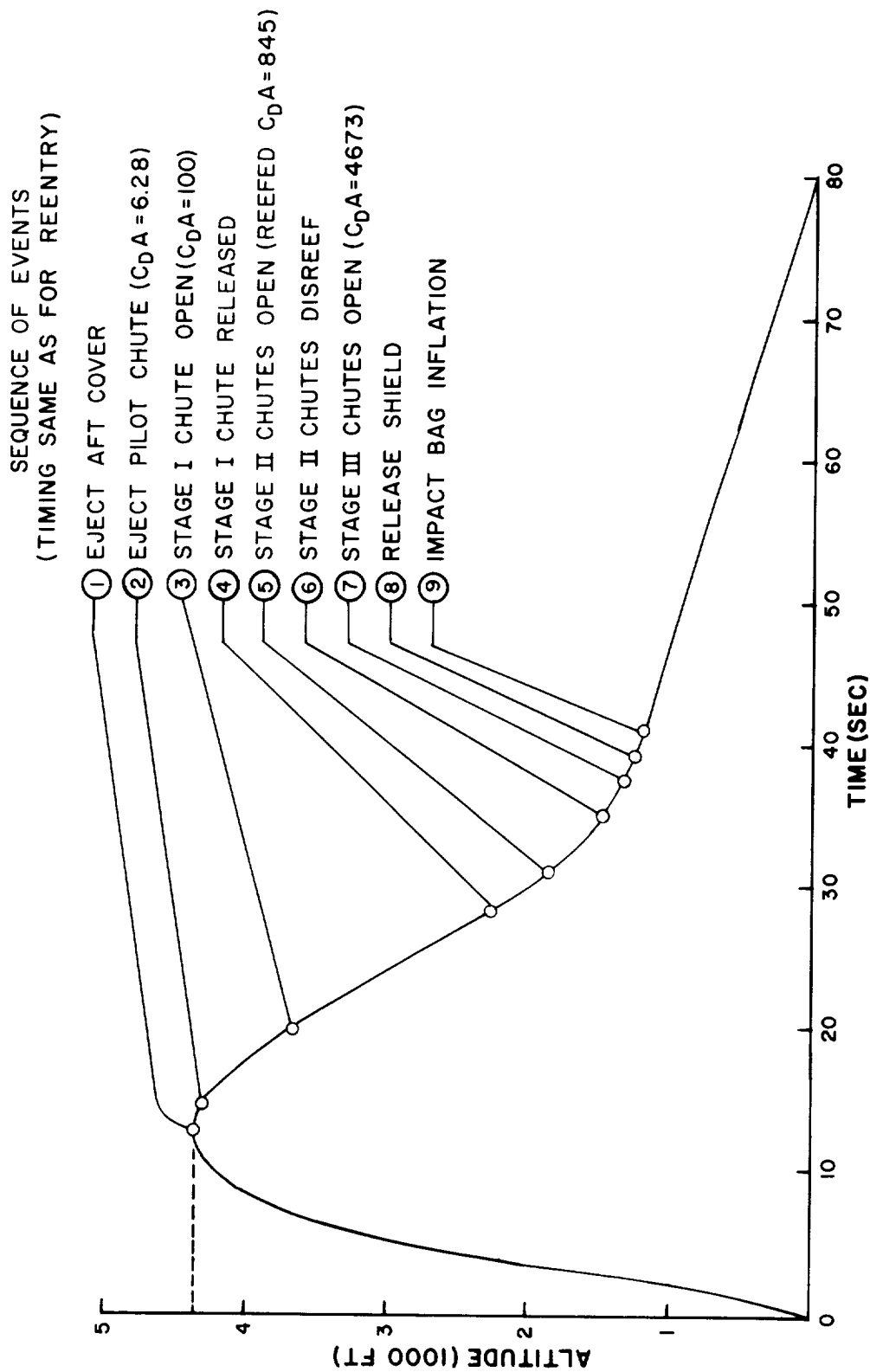


Figure III-5-11. Launch pad abort sequence
D-2 Configuration

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Subsequent possible abort conditions from normal missile lift-off will reflect a family of ballistic trajectories, each increasing in apogee altitude as stage I powered flight progresses. This requires that the 12.5-second timing interval established for the launch pad abort case be extended in order to prevent initiating the recovery sequence as the re-entry vehicle is on the upward side of the trajectory approaching apogee. Once an apogee exceeding 25,000 feet (or the setting of the pressure sensor) can be guaranteed, the timed signal previously used for initiating the first recovery event can be used for arming the pressure sensor and the sequence will be automatically initiated at the minimum fixed altitude on the downward side of the trajectory for all possible ballistic trajectories during later phases of powered flight.

Taking as an example a case of abort just prior to stage I burnout as shown in Figure III-5-12, separation of the re-entry vehicle would activate the initial time delay. The re-entry vehicle is then at an altitude exceeding 72,000 feet and arming of the altitude sensor could take place after the timing interval had elapsed. The re-entry vehicle would then be on a ballistic flight path in which ambient pressures would at all times be less than the set pressure of the altitude sensing device until it descended to 25,000 feet. The recovery sequence would then be initiated and a similar condition would exist for subsequent abort trajectories during later stages of powered flight.

In addition to providing automatic control of the lower operational limit, namely minimum altitude, arming would also be provided for the upper limiting conditions of high velocity or high dynamic pressures. Where broad band operational conditions for the recovery system exist such as long-range ballistic abort flights or return from orbit, arming of recovery initiation will be provided in series with the altitude sensing device. A reference for such arming will be established by "g" plus time from actual flight performance. This will permit the crew to exercise optional control of the recovery sequence between the permissible limits of parachute deployment. It would also provide a safety feature in case of altitude sensing malfunction.

In the case of an abort condition just prior to second-stage burnout as shown in Figure III-5-13, the high dynamic pressure (or deceleration) transition could be used as a reference to arm for maximum dynamic pressures. In this case arming for minimum altitude recovery initiation would take place prior to the high q transition but the actual firing circuit for the first recovery event would remain unarmed until the "g" transition had been completed. The crew would then be able to exercise manual control of recovery sequence initiation between the altitudes of 80,000 feet and 25,000 feet without exceeding the design limitations of the parachute subsystem.

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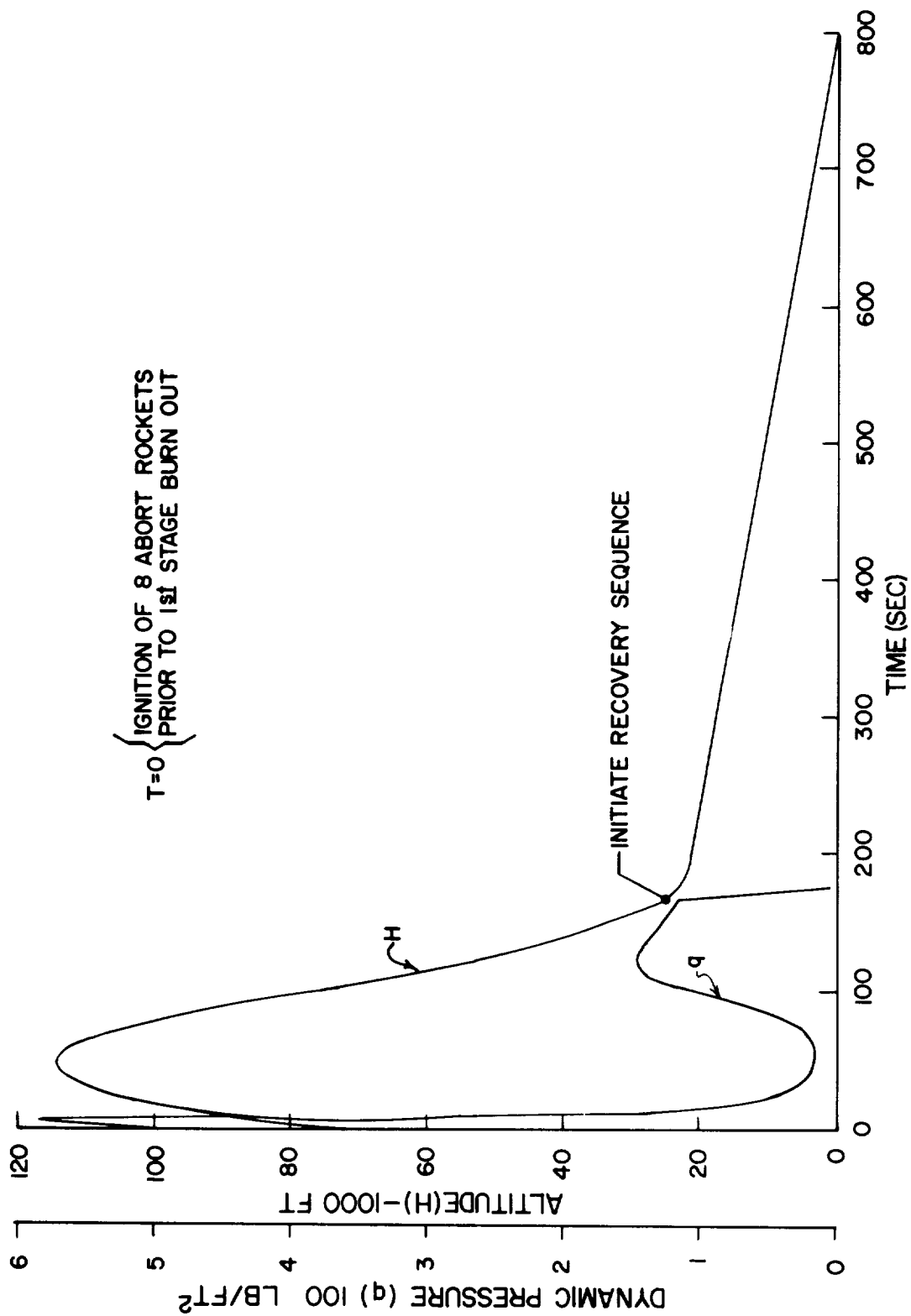


Figure III-5-12. First stage powered flight abort, D-2 Configuration

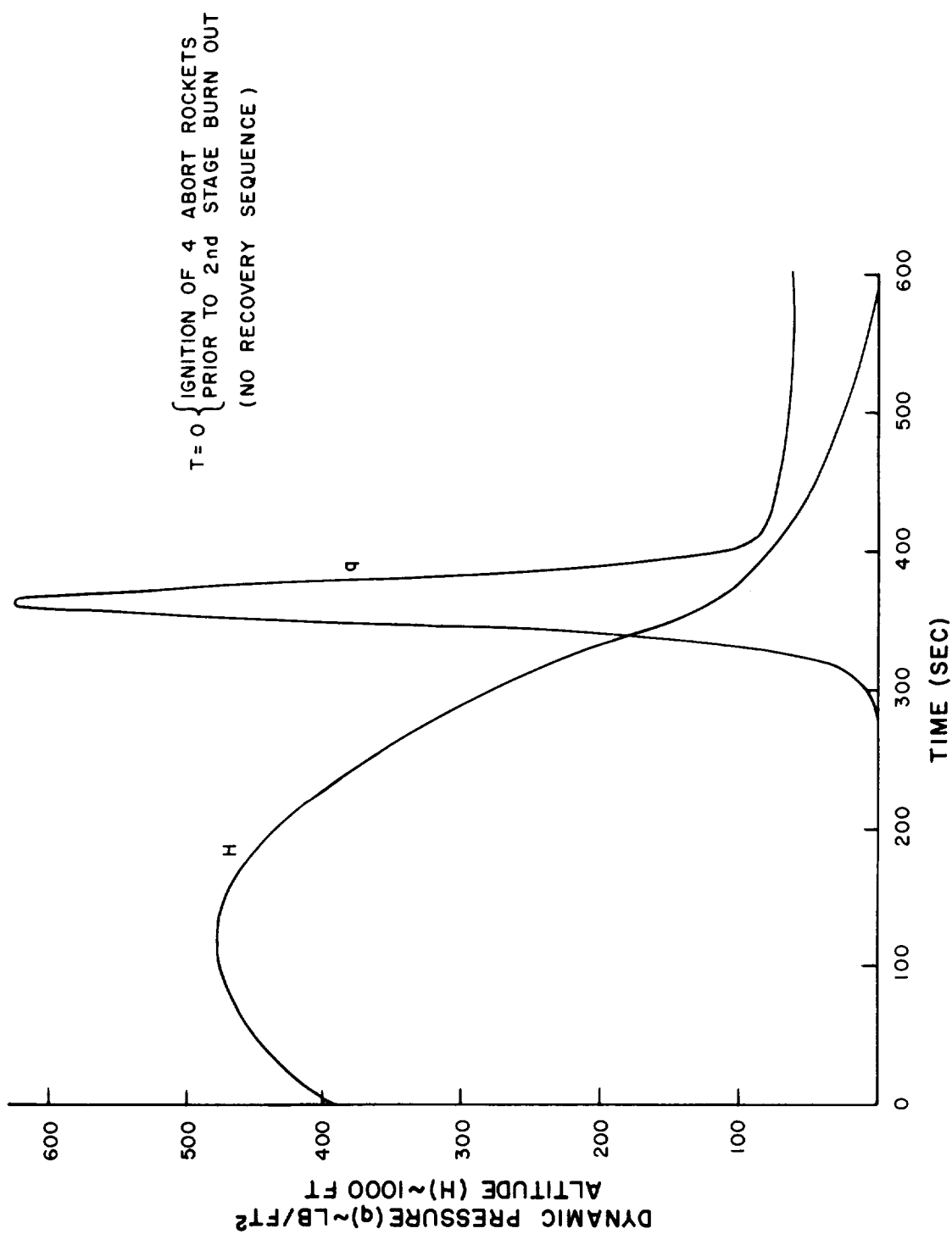


Figure III-5-13. Second stage powered flight abort, D-2 Configuration

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Summarizing the above, the following means of arming, sensing and control appear feasible for the D-2 configuration recovery system.

| <u>CONDITION</u> | <u>ARMING</u> | <u>SENSING</u> | <u>CONTROL</u> |
|--|-------------------------------|-----------------|----------------------|
| I Off the pad abort | R/V Separation | Timer | Automatic |
| II First stage powered flight (immediately after launch to t = 80 sec) | R/V Separation | Timer | Automatic |
| III First stage powered flight (t = 80 sec to burnout) | R/V Separation | Pressure Sensor | Automatic |
| IV Second and third stage powered flight | R/V Separation and g's + time | Pressure Sensor | Automatic and Manual |
| V Return from earth or lunar orbit | R/V Separation and g's + time | Pressure Sensor | Automatic and Manual |

5.5.2 Retardation System

A large number of launch pad abort trajectories has been calculated for various launch attitudes, accelerations, and rocket burning times. An analysis of the launch pad abort from the entire APOLLO system viewpoint resulted in the selection of a group of abort trajectories which satisfy the APOLLO system requirements. This group was then analyzed from the recovery system viewpoint.

The recovery system selected is indicated in Volume VI as retardation system no. VI and consists of a 25 foot diameter First ribbon chute 1st stage and three 53 foot diameter main chutes. The main chutes are initially deployed in a reefed condition of 19 feet dia to reduce the opening shock.

The systems analyzed in Volume VI were based on abort data available at the time of the mid-term review. As previously stated, abort system requirements were changed to reflect the latest available data on the Saturn booster system. Therefore, the parachute deployment sequence detailed in Volume VI was modified for use with the latest abort trajectories. The revised sequence is now identical to the time sequence used for normal re-entry recovery including the inclusion of 1st stage chute drag time.

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The sequence of operation, with $t = 0$ being the deployment initiation command, is presented in Table III-5-I.

TABLE III-5-I PARACHUTE DEPLOYMENT SEQUENCE

| <u>Time (sec)</u> | <u>Event</u> |
|-------------------|--|
| 0 | Eject aft cover. No drag effects on R/V |
| 2.0 | Eject pilot chute (4' ribbon). No drag effects on R/V |
| 6.0 | First stage chute extended. No drag effects on R/V |
| 7.2 | First stage chute fully open (25' First ribbon). $(C_D A)_{\text{Total}} = 182$ (including $(C_D A)_{R/V} = 82$) |
| 15.2 | Extend main chutes at completion of 8 sec. drag time. |
| 18.5 | Main chutes (3) open reefed (19' dia.). $(C_D A)_{\text{Total}} = 927$ |
| 22.5 | Actuate reefing cutters |
| 25.0 | Main chutes fully open (53-ft dial) $(C_D A)_{\text{Total}} = 4673$ |

5.5.3 Obstacle Avoidance

The attractive weight, packaging and reliability aspects of parachutes make them the optimum selection as recovery devices at the present time. However, conventional parachutes have two undesirable features as a recovery device in that they possess:

Little capability to maneuver and avoid obstacles on landing such as rocks, trees, power lines, etc.

Little capability of correcting for wind drift.

The importance of the first feature above can be appreciated if the effects on impact attenuation caused by the above mentioned obstacles are considered. The second undesirable parachute characteristic, wind drift, can also affect the operation of the impact attenuation devices by imparting a horizontal velocity to the vehicle which is undesirable

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for certain impact attenuation devices. Wind can also cause drift back over the pad on launch pad abort missions. It is, therefore, clear that the parachute recovery system would become even more attractive if some means were available to overcome wind drift and provide last minute maneuvering capability. One method of handling such a situation is to modify a conventional parachute so that it will be capable of glide maneuvers with preferential orientation. This type of parachute is referred to as a controllable or dirigible parachute. Two investigators presently working with this type of parachute are the Radioplane Division of Northrop Corporation and the Sandia Corporation, references 5-23 and 5-24.

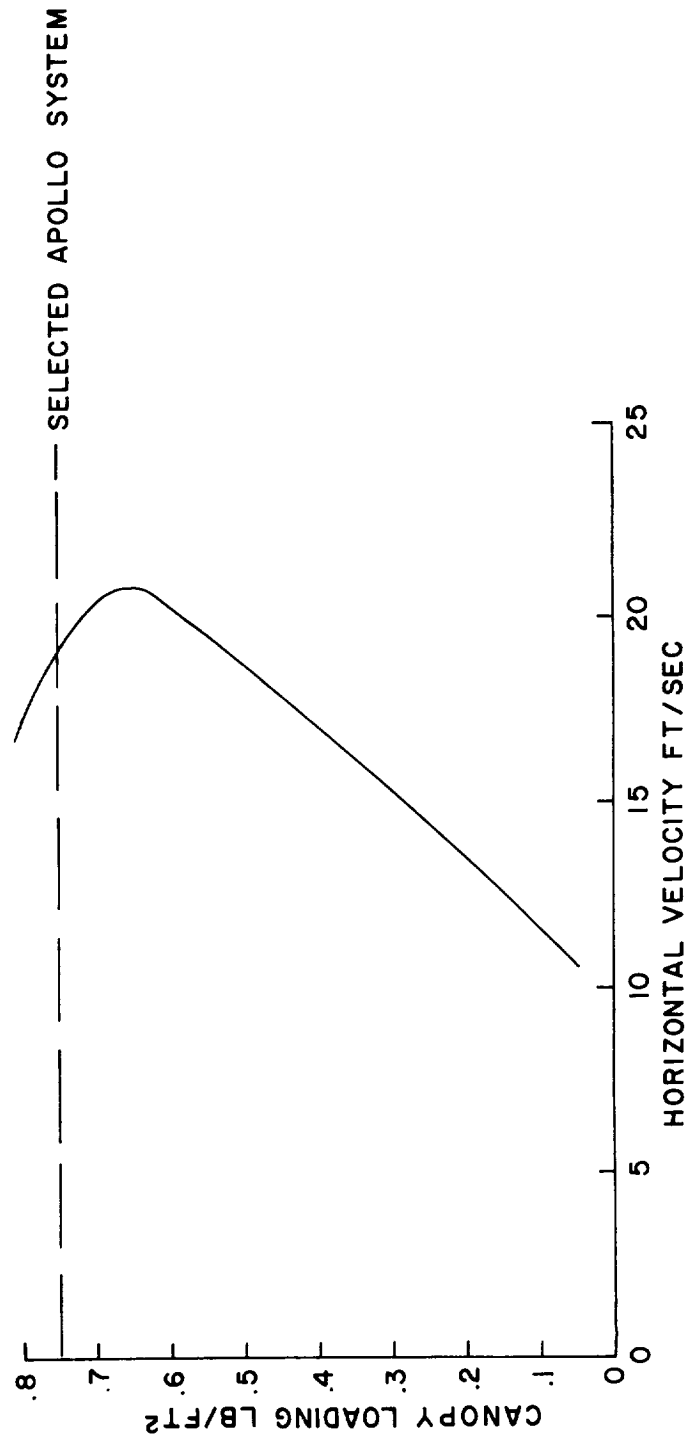
The Sandia system is a conventional parachute with a gore or gores removed to provide the thrust for gliding. The directional control is obtained by pulling on a guide line which is attached to the parachute skirt at the corner of the removed gore, thereby giving rotational control in two directions. Wind tunnel and drop tests have shown that glide angle and rotational response were good for a small solid, flat circular chute. Figure III-5-14 shows test results obtained with this chute system with the canopy loading of the selected APOLLO system superimposed. It can be seen that the selected system has almost the optimum loading value shown for maximum glide angle.

The Radioplane dirigible parachute is the second type being investigated. This type of controllable parachute has a section of the canopy which is hinged to the main canopy near its vertex. For normal descent it is held in its normal position by control lines. For initiation of the glide the control lines are played out allowing the skirt of the hinged section to rise. This produces a "hole" in the side of the chute producing a thrust and, therefore, gliding. Directional control is achieved by varying the length of the control lines so that the skirt of the hinged section is pulled down on one side. This produces a twist of the hinged section which changes the direction of the thrust vector so that rotation of the chute is obtained. Qualitative drop tests have been made on single and clustered parachutes of this type. Typical results obtained were:

-
- Ref 5-23 Ewing, E. G., Radioplane Dirigible Parachute Development. Radioplane Report No. PTM-332, Feb. 1961
- Ref 5-24 Kane, M. T., A Guided Parachute System. Proceedings of the Recovery of Space Vehicle Symposium, Institute of the Aeronautical Sciences, August, 1960.

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Figure III-5-14. Effect of canopy loading on horizontal velocity

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$$V_{\text{horizontal}} = 8.5 \text{ ft/sec}$$

$$V_{\text{vertical}} = 15.8 \text{ ft/sec}$$

$$\text{Average glide angle} = 28 \text{ degrees}$$

for a conical type approximately 40 feet in diameter.

5.5.3 Modified Lenticular Re-entry Vehicle

The R-3 configuration vehicle utilizes its aerodynamic maneuvering characteristics to effect a conventional landing for abort as well as normal re-entry recovery. The system utilized is detailed in Volume VI. In the event a conventional landing cannot be effected, or if a water recovery is required, a backup parachute system is included for this vehicle. As shown in Figure III-5-15, a single 43 foot ring-sail chute is utilized to provide terminal velocity of 70 ft/sec with a maximum side drift of 30 ft/sec.

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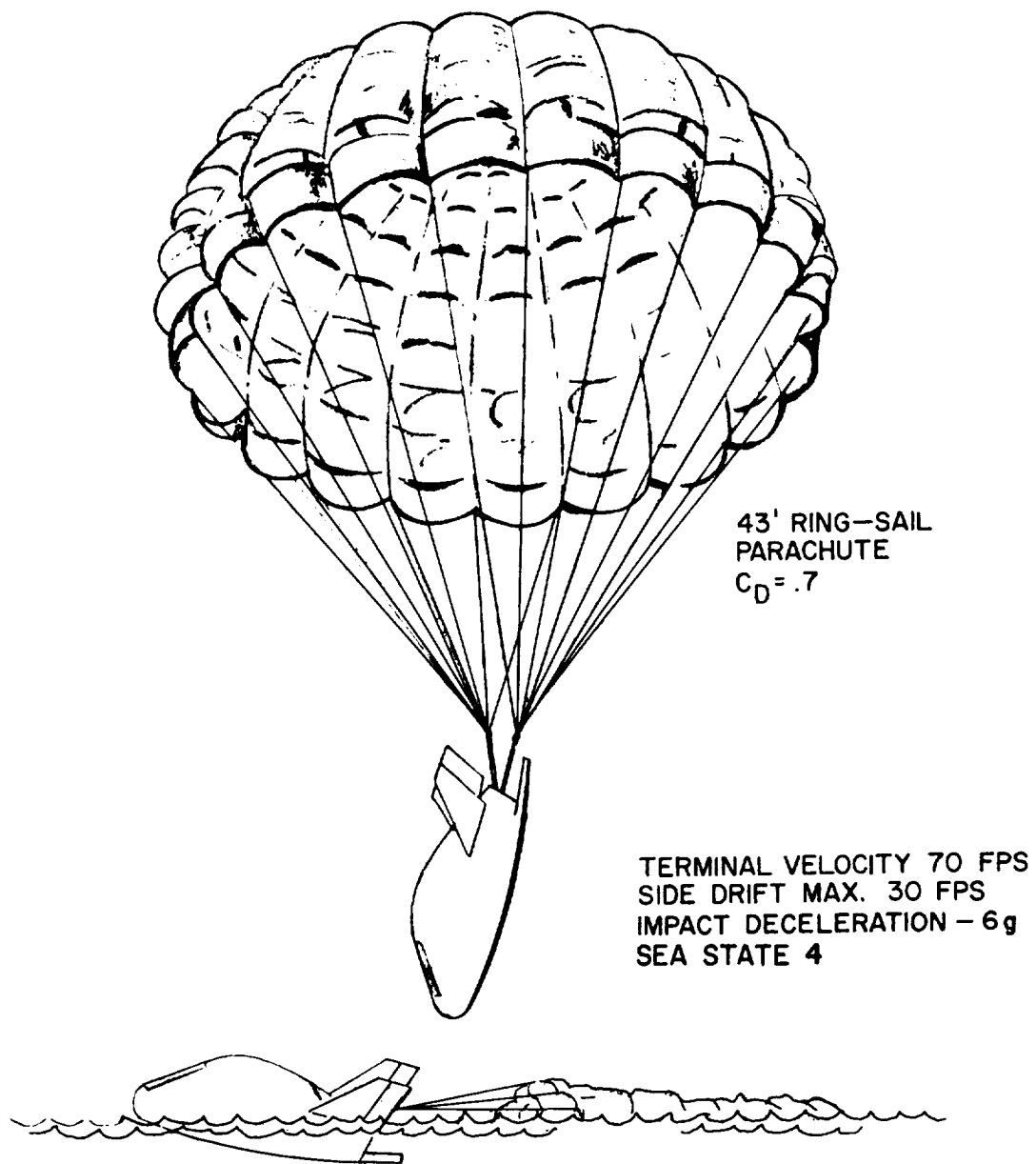


Figure III-5-15. Emergency water recovery, R-3 Configuration

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6.0 Recommended Study Areas for Further Work

During the study program it became obvious that certain study areas could not be completely covered during a six-month program. Other areas, in which sufficient data was found lacking or non-existent were also revealed.

It is recommended that, as a sequel to this six-month study and to insure that the required knowledge is available for the development program, work be continued in the areas discussed below.

6.1 LENTICULAR VEHICLE FLIGHT TESTS

The NASA Flight Research Center, Edwards Air Force Base, California is presently conducting a test program to simulate the launch pad abort of the Dynasoar vehicle. An F5D aircraft, which has approximately the same wing loading and is capable of matching the L/D and C_L vs. α curves of the Dynasoar vehicle, is utilized in this program. By varying the drag (by use of flaps or landing gear) the L/D range can be extended to cover the R-3 modified lenticular vehicle regime. With this in mind, specific information relative to the R-3 vehicle was requested by NASA FRC in order that they might determine the possibility of simulating the R-3 vehicle configuration in a series of these tests.

It is recommended that this program be conducted as soon as possible so as to further evaluate the capabilities of the modified lenticular vehicle.

6.2 BOOSTER EXPLOSION

Early in the course of this study program it became apparent that very little data is available relative to the explosive effects of large multistage boosters. In addition, the explosive yield of specific fuels, when stored in quantity in such boosters, has not been firmly established. The figure of 50 percent of the weight of the total fuel in equivalent TNT utilized in this program was obtained with precautionary advice that, although it was the best number available at the time (April 13, 1961), it was subject to change and would probably have to be increased when further data became available.

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As can be seen from the calculations of Sections 4.1.1 and 5.1, the accelerations required to propel the abort vehicle away from an exploding booster approach the intolerable level even when based on this 50 percent figure. In the event the estimate of explosive yield is increased it will be necessary to provide additional warning time in order that the acceleration profile not be increased.

It is therefore recommended that a two part study be actively pursued in order to

(a) determine, within reasonable tolerance limits, the explosive yield of multistage boosters with tank capacities ranging upwards from the Atlas to well beyond the Saturn C-2, and

(b) investigate methods of providing greater minimum warning time for impending explosions.

This latter should be carried through the preliminary design and initial hardware stages as expeditiously as possible as it may well be a limiting item for manned space flight utilizing large boosters.

6.3 ABORT TRAJECTORY OPTIMIZATION

The problem of minimum warning time for launch pad abort should also be investigated with the objective of determining the minimum warning time requirement — as well as the minimum warning time capability. The latter, as outlined in the previous section, is a function of booster system characteristics and warning system capabilities. The former, although parametrically studied during this program (see Section 4.1), can be further optimized by erecting a mathematical model of the launch abort system with specified limits on the critical parameters. As indicated in the following paragraphs, work along this line has already been started at the General Electric Co. and should be continued beyond the completion of this study program.

6.3.1 Analysis of the Abort Trajectory

The problem of ejecting an abort vehicle under launch conditions will be considered by erecting a mathematical model representing physical reality within meaningful engineering limits. In order to write the dynamic and kinematic equations of motion the following hypotheses have been introduced:

(a) the vehicle is considered as a mass-point system, therefore the moment equation about the c.g. of the vehicle is disregarded.

(b) the thrust vector is coincident with the velocity vector at any point on the abort trajectory.

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- (c) the vehicle is non-lifting.
- (d) the thrust is constant.

In the light of the above considerations, the dynamic and kinematic equations of motion, referred to a fixed cartesian coordinate system x, y , (see Figure III-6-1) are written as follows:

$$\psi_1 = Z' + \sigma' \bar{V}_e + D^*(z, \tilde{h}, L^*) \exp(-\sigma) + \alpha = 0 \quad (1)$$

$$\psi_2 = \alpha' + \left[1 - \alpha^2 - L^* \exp(-\sigma) (1 - \alpha^2)^{1/2} \right] Z^{-1} = 0 \quad (2)$$

$$\psi_3 = \xi' - Z (1 - \alpha^2)^{1/2} = 0 \quad (3)$$

$$\psi_4 = \tilde{h}' - Z \alpha = 0 \quad (4)$$

where the following replacements and non-dimensional variables have been introduced.

$$\begin{aligned} Z &= \frac{V}{V_R}, & \sigma &= Lg \tilde{m}, & \tilde{m} &= \frac{m}{m_0}, & \bar{V}_e &= \frac{V_e}{V_R} \\ \alpha &= \sin \theta, & D^* &= \frac{D}{m_0 g}, & L^* &= \frac{L}{m_0 g}, & \xi &= \frac{xg}{V_R^2} \\ \tilde{h} &= \frac{hg}{V_R^2}, & \tau &= \frac{tg}{V_R}, & \eta &= \frac{\rho}{\rho_R} = \eta(\tilde{h}) \end{aligned}$$

The aerodynamic drag of the configuration is expressed as:

$$D^* = k \eta(\tilde{h}) Z^2 C_{D_0}(M) + \frac{K(M)}{k \eta(\tilde{h}) Z^2} (L^*)^2 \quad (5)$$

where

$$k = \frac{1}{2} \frac{\rho_0 V_R^2 S}{m_0 g}, \quad M = Z \frac{V_R}{a_R} \phi^{-1/2} = M(Z, \tilde{h})$$

Thus,

$$\phi = T/T_R$$

$$D^* = D(Z, \tilde{h}, L^*) \quad (6)$$

The set of differential equations (1) to (4) has two degrees of freedom associated with the choice variables $L^*(\tau), \sigma(\tau)$ therefore it is possible to impose an optimization requirement. In other words, the variational question of finding the optimum histories of $L^*(\tau), \sigma(\tau)$ minimizing the increment of a functional $H = H(Z, \alpha, \xi, \tilde{h}, \sigma, \tau)$ between the terminal points,

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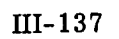


Figure III-6-1. Optimization of abort study

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may be posed. The physical problems associated with the optimization of the argument functions $L^*(\tau)$ and $\sigma(\tau)$ are called "optimum lift program" and "optimum thrust program", respectively.

For the problem being considered here it is assumed that a non-lifting body ($L^*=0$) is propelled by a constant thrust rocket engine. From the common expression for the thrust of a rocket engine, disregarding the contribution of the pressure terms, one obtains:

$$T = - \frac{dm}{dt} V_e = -\dot{m} V_e \quad (7)$$

Therefore, for the case where $T = \text{const.}$, it is found that:

$$\sigma' = \frac{d\sigma}{d\tau} = - \frac{V_R}{g} \frac{\frac{c^*}{c^* V_R} \tau}{1 - \frac{c^* V_R}{g} \tau} = \sigma'(\tau) \quad (8)$$

$$\sigma = Lg \left(1 - \frac{c^* V_R}{g} \tau \right) = \sigma(\tau) \quad (9)$$

Notice that:

$$\tilde{m} = 1 - \frac{c^* V_R}{g} \tau \quad (10)$$

As a consequence of the preceding considerations, the system of differential equations (1) to (4) has no degree of freedom in this case, ($L^*=0$, $T = \text{const.}$) Nevertheless, an optimization criteria may still be imposed. The problem, though it is no longer approached by variational techniques, shifts to the realm of the ordinary maxima and minima as an optimization of the boundary conditions.

The symbols used in this Section (6.3) are as follows:

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SYMBOLS

| | |
|--------------------------------------|--|
| Z | Non-dimensional velocity |
| ξ | Non-dimensional range |
| \tilde{h} | Non-dimensional altitude |
| V | Flight velocity |
| V_e | Velocity of the gases at the exit section of the rocket engine |
| D^* | Non-dimensional drag |
| L^* | Non-dimensional lift |
| ρ | Density |
| V_R | Reference velocity |
| m | Mass of the capsule |
| a | Speed of sound |
| a_R | Reference speed of sound |
| g | Acceleration of gravity |
| τ | Non-dimensional time |
| ϕ | Non-dimensional temperature |
| S | Reference surface |
| T | Absolute temperature |
| c^* | Ratio of mass flow to initial mass |
| \tilde{m} | Non-dimensional mass |
| τ_R | Time for the shock front to reach the safe distance r . |
| τ_W | Warning time |
| M | Mach number |
| $\frac{d}{dt} (\dots) = (\dots)$ | |
| $\frac{d}{d\tau} (\dots) = (\dots)'$ | |

SUBSCRIPTS

| | |
|-----|---|
| R | Indicates a reference value |
| o | Indicates a zero-lift condition or an initial value |

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6.3.2 An Optimization Criteria for the Abort Mission

A possible malfunction of the booster system at launch might cause an explosion endangering the lives of the crew. In such a case it is desirable to eject the abort vehicle as soon as the malfunction is detected in order to maintain a safe distance between the vehicle and the center of explosion at the proper time. With reference to Figure III-6-1, the explosion, with center at C, originates a circular shock front F which is assumed to propagate circularly and concentrically in time. Through the shock front the static pressure rise Δp decreases in time, as the radius r increases. Once the pressure jump Δp reaches a value of 10 p.s.i. it is assumed that, on the basis of structural limitations of the vehicle, the safe distance has been attained. The time and distance traveled by the shock wave to the point where Δp attains the specified value is calculated from a study of shock propagation as indicated in Section 5.1.

Along the abort trajectory the abort vehicle position may be identified by the polar coordinates (ρ, β) . With respect to the fixed system (x, y) the shock wave circular front may be written:

$$f(x, y) = x^2 + y^2 + 2yd + d^2 - r^2 = 0 \quad (11)$$

where d and r are given values.

The optimization criteria formulated is to find the optimum c^* (optimum thrust) and α_0 (optimum steering) minimizing the time $\tau_{\rho\xi}$ for the capsule to reach a distance $\rho = (x^2 + y^2)^{1/2}$ where x and y satisfy the condition $f(x, y) = 0$, τ_{ρ} being less or equal than $\tau_r + \tau_w$.

The optimum problem defined may be called an "open" question. In other words, since other requirements may be imposed because of engineering considerations on ulterior phases of the abort mission, the class of trajectories may be narrowed. For example, the success of the recovery depends on a parachute sequence which requires an apogee, to be attained in the abort path, within certain values, $\tilde{h}_A \leq \tilde{h}_B \leq \tilde{h}_C$. Therefore, in such case the problem will be a "conditioned" optimum. Other conditions may be imposed and analyzed as well; e.g., given range at the apogee, etc.

To study the above problem a set of solutions of the differential equations (1) to (4) is needed. In general at $f=0$

$$\tau_{f=0} = \tau(Z_0, \alpha_0, \xi_0, \tilde{h}_0, c^*) \quad (12)$$

for abort trajectories with $L^* = 0$, $\tau_0 = 0$.

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For given initial conditions $Z_0 = 0$, $\xi_0 = 0$, $\tilde{h}_0 = 0$,

$$\tau_{f=0} = \tau(\alpha_0, C^*) \quad (13)$$

Therefore the optimizing conditions may be obtained from a set of holonomic equations,

$$\begin{aligned} \Psi_1 &= \Psi_1(\alpha_0, C^*) = 0 \\ \Psi_2 &= \Psi_2(\alpha_0, C^*) = 0 \end{aligned} \quad (14)$$

derived from $\tau_{\alpha_0} = \tau_{C^*} = 0$

Due to the tri-dimensional character of the solution the problem requires the integration of a series of trajectories and a careful analysis using level curves. The graphical plotting of the results is necessary in order to make additional studies once the other requirements are imposed on the solution. An analytical expression of the function (13) is also desirable.

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